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Journal

of

THE ROYAL AERONAUTICAL SOCIETY

SEPTEMBER 1958

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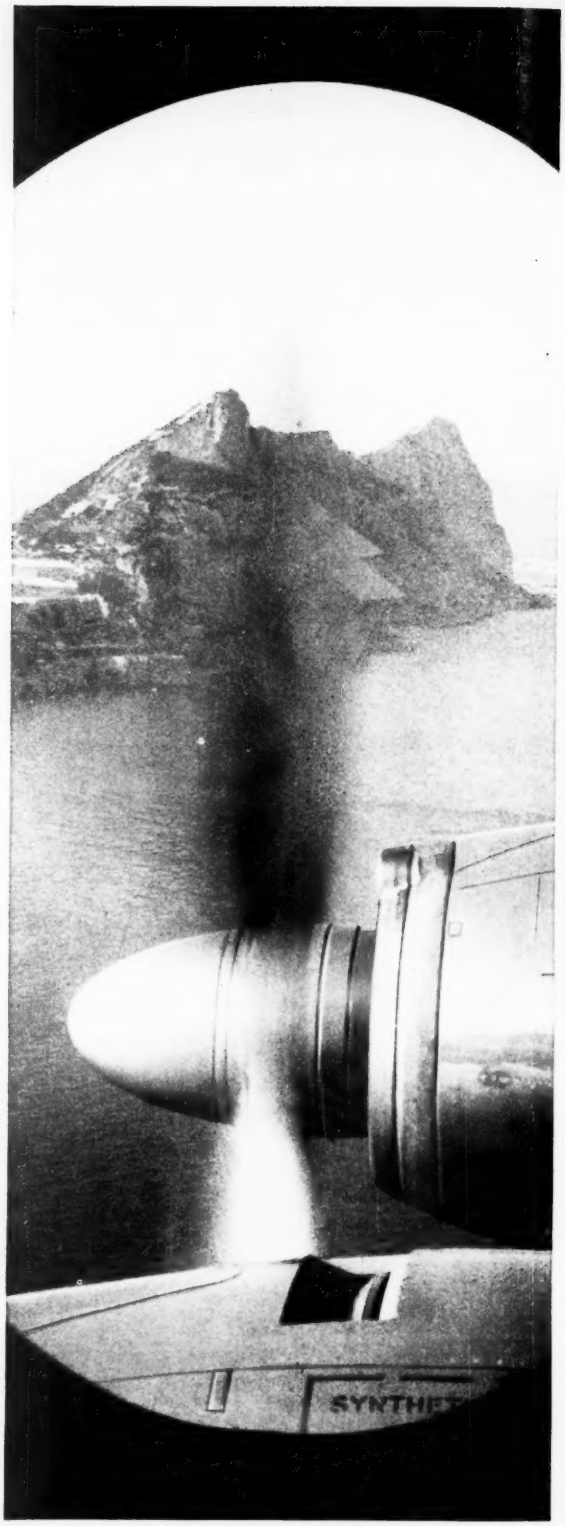

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All communications for publication in, and advertisements for, the JOURNAL *should be addressed to the Editor,* JOAN BRADBROOKE, *A.R.Ae.S.; all communications on general matters affecting the Society should be addressed to the Secretary,* A. M. BALLANTYNE, *T.D., B.Sc., Ph.D., Hon.F.C.A.I., A.F.I.A.S., F.R.Ae.S.*

None of the papers or paragraphs must be taken as expressing the opinion of the Council of the Royal Aeronautical Society unless otherwise stated.

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NOTICES

The Fourteenth British Commonwealth Lecture

The Fourteenth British Commonwealth Lecture will be delivered on Thursday 9th October 1958 by Mr. J. C. Floyd, F.R.Ae.S., F.C.A.I., M.I.A.S. Mr. Floyd is the Vice-President (Engineering) of Avro Aircraft Limited, Canada, and his subject will be "Some Aspects of Canadian All-Weather Fighter Development."

The lecture will be held at the Royal Institution, 21 Albemarle Street, London, W.1, at 6 p.m. (Tea will be served at 5.30 p.m.)

International Congress—Madrid

The programme of papers to be presented at the First International Congress of the Aeronautical Sciences, 8th-13th September, in Madrid, was published in the July JOURNAL. The Institute of the Aeronautical Sciences, the permanent Secretariat of the International Council of the Aeronautical Sciences, has appointed the Pergamon Press Ltd. publishers for the Congress papers. They will publish a Proceedings and are in charge of all advance copies of the 45 papers to be given at Madrid.

Advance copies of the papers, which are obtainable singly, will be available in Madrid and may also be obtained at the beginning of September from Pergamon Press Ltd., Pergamon House, 4 Fitzroy Square, London W.1, at 3s. 6d. per copy.

Prizes in Aeronautics

The Society's Prizes in Aeronautics, awarded annually to the best student in the Aeronautical Department at Universities and Colleges organising courses in aeronautics, have been awarded for 1958 as follows:—

University of Bristol—Joint Award—A. G. Fraser and D. H. Lloyd.

Cambridge University—R. A. Sawyer.

Glasgow University—D. B. Gemmell.

Imperial College, London—Colin Wood.

Queen Mary College, London—J. F. A. Nash.

Southampton University—J. A. Wilson.

College of Aeronautics, Cranfield—A. G. J. Baker.

Elliott Memorial Prize

The Elliott Memorial Prize has been awarded to Sergeant Apprentice P. J. Parry, the Aircraft Apprentice at R.A.F. Halton, who gained the highest marks in the General Studies Examination. The Prize was awarded to him at the Passing Out Parade in July.

The Glasgow Branch

The Glasgow Branch of the Society has formed a Graduates' and Students' Section which is being organised by a "steering" Committee of eight members under the Chairmanship of Dr. A. W. Babister, A.F.R.Ae.S., Chairman of the Branch. Honorary Secretary of the Section is Gordon H. K. Goold, Grad.R.Ae.S. An interesting programme of lectures and visits has been arranged for the Section and an enthusiastic response is expected from Glasgow University, the Technical Colleges and Industry.

Representatives on Other Bodies

Dr. A. M. Ballantyne has been appointed the Society's Representative from 31st August 1958 on the Regional Advisory Council for Higher Technological Education in succession to Sir John Buchanan who has so ably represented the Society since the inception of the Advisory Council.

Guided Flight Section

The Annual General Meeting of the Guided Flight Section of the Society will be held on Tuesday 16th September at the offices of the Society, 4 Hamilton Place, London, W.1, at 5.30 p.m. Tea will be served after the meeting.

Royal Aeronautical Society Scholarships and Grants 1958/1959

The following Scholarships and Grants have been awarded by the Council:—

The Charter Scholarship—to J. A. Wilson of St. Andrew's Villa, Heckington, Sleaford, Lincs., for research on Hypersonics. He is 22 years old and holds the Degree of B.Sc.(Eng.) with First Class Honours of Southampton University, where he is to continue his studies.

The Geoffrey de Havilland Memorial Scholarship—to A. J. W. Smith of 352 Woodham Lane, New Haw, Weybridge, for post graduate work on aerodynamics and aircraft design at the College of Aeronautics, Cranfield. He is 21 years old and holds the Higher National Diploma in Mechanical Engineering.

Educational Grants have been awarded to:—

J. M. Moten to enable him to continue the post graduate course at the College of Aeronautics, Cranfield. He is 24 years old and holds the degree of B.Sc. of Adelaide University and the degree of Bachelor of Aeronautical Engineering of Sydney University. He is specialising in Electrical Engineering and Aerodynamics.

R. W. Roberts of 67 Cadogan Gardens, London, S.W.3, to enable him to continue his work on secondary flow in rotating channels. He is 29 years old, and holds the degree of B.Sc.(Mech.Eng.) of the University of Cape Town, and the degree of M.Sc.(Thermodynamics) of the University of Birmingham.

(The Edward Busk Studentship in Aeronautics was not awarded this year.)

The Dawn of Aerodynamics

In Captain Pritchard's paper "The Dawn of Aerodynamics" (JOURNAL, March 1957), the third reference is to "An account of a book entitled *New Principles of Gunnery* . . . by Benjamin Robins, *Phil. Trans.*, 14th and 21st April 1743." The Library of the Society has now acquired a copy of the original book.

It is interesting to note that Professor R. S. Hartenberg of Northwestern University, Illinois, delivered a lecture earlier this year on "Benjamin Robins: Founder of Experimental Aerodynamics."

The Institution of Chemical Engineers, The Society of Instrument Technology and The British Computer Society Symposium

A Joint Symposium on Instrumentation and Computation in Process Development and Plant Design, will be held in the Central Hall, Westminster, on the 11th, 12th and 13th May 1959, with subjects and sessions as follows:

11th May—Morning Session
Improving the efficiency of existing processes.

11th May—Afternoon Session
The design of new processes.

12th May—Morning Session
The application of on-line computers.

12th May—Afternoon Session
Recent developments in instruments, on-line computers and computers for design.

13th May—Morning Session
(*a*) The use of computer techniques in large and small companies.
(*b*) The future (one paper only).

Continued

This meeting is being organised by The Institution of Chemical Engineers, The Society of Instrument Technology and the British Computer Society, under the aegis of The British Conference on Automation and Computation. Any member wishing to attend should contact The General Secretary, The Institution of Chemical Engineers, 16 Belgrave Square, London S.W.1, for further details.

ACKNOWLEDGMENTS

The Council wishes to thank Scandinavian Airlines System for presenting the Society with a first flight cover carried on the first S.A.S. regular flight Stockholm-Djakarta in January this year; and Air France for presenting a cover carried on its first regular service Paris-Tokyo, by way of the North Pole, on 10th April 1958.

These two covers are important additions to the Society's collection of air mail covers.

DIARY

LONDON

16th September

GUIDED FLIGHT SECTION.—Annual General Meeting. 4 Hamilton Place. 5.30 p.m.

9th October

THE FOURTEENTH BRITISH COMMONWEALTH LECTURE.—Some Aspects of Canadian All-Weather Fighter Development. J. C. Floyd. Royal Institution, 21 Albemarle Street. 6 p.m. (Tea at 5.30 p.m.)

27th October

LECTURE*—The Bloodhound. D. J. Farrar. Institution of Civil Engineers, Great George Street, S.W.1. 6 p.m. (Tea at 5.30 p.m.)

*Arranged by the Committee of the Guided Flight Section.

GRADUATES' AND STUDENTS' SECTION

25th September

The Next Generation of Civil Aircraft. G. H. Lee. Library, 4 Hamilton Place. 7.30 p.m.

16th October

The Fatigue of Aircraft. Major P. L. Teed. Library, 4 Hamilton Place. 7.30 p.m.

29th October

Annual Film Show. Library, 4 Hamilton Place. 7.30 p.m.

BRANCHES

15th September

Halton.—Physiological Aspects of High Speed Flight. Sq. Ldr. T. C. D. Whiteside. Branch Hut, R.A.F. Halton. 6.45 p.m.

17th September

Coventry.—Design of Pattern Equipment for an Aero Engine Casting. K. Marston. Wine Lodge, Coventry. 7.30 p.m.

Weybridge.—The Tyne Engine. D. McLean. Apprentice Training School, Vickers-Armstrongs (Aircraft) Ltd. 6.10 p.m.

19th September

Birmingham.—Gliding. M. S. Hunt. The Birmingham Exchange and Engineering Centre, Stephenson Place. 7.30 p.m.

22nd September

Henlow.—Annual General Meeting and Film Show. Building 62, R.A.F. Technical College, Henlow. 7.30 p.m.

23rd September

Luton.—Film Evening. Napier Senior Staff Canteen, Luton Airport. 6.15 p.m.

25th September

Isle of Wight.—Artificial Satellites, The Present Position and Future Possibilities. Dr. L. R. Shepherd. Clubhouse, Saunders-Roe Sports and Social Club, E. Cowes. 6.30 p.m.

30th September

London Airport.—The Boeing 707. M. Pennell (Boeing Airplane Co.). B.O.A.C. Headquarters, L.A.P. 6 p.m.

3rd October

Luton.—Buffet Dance and Social Evening. Napier Senior Staff Canteen, Luton Airport. 6.15 p.m.

6th October

Henlow.—Mixed Power Plants. M. J. Brennan. Building 62, R.A.F. Technical College, Henlow. 7.30 p.m.

7th October

Boscombe Down.—Propulsion Systems for Space Travel. A. V. Cleaver. A. & A.E.E. Lecture Hall. 5.45 p.m.

8th October

Gloucester.—The History of the Supermarine Works. A. N. Clifton. Wheatstone Hall, Gloucester. 7.30 p.m.

Swindon.—The Work of the R.A.E. M. B. Morgan. The College, Victoria Road. 7 p.m.

Weybridge.—The Physiological Aspect of High Performance Flight. Dr. K. G. Williams. Apprentice Training School, Vickers-Armstrongs (Aircraft) Ltd. 6.10 p.m.

13th October

Glasgow.—Graduates' and Students' Section Opening Meeting—Chairman's Address and Film Show. College of Science and Technology, Duke Street. 7.30 p.m.

15th October

Bristol.—Presidential Address. Filton House. 6 p.m.

Coventry.—Digital Computers. R. F. King. Wine Lodge, Coventry. 7.30 p.m.

20th October

Glasgow.—Propulsion Systems for Helicopters. Dr. A. W. Morley. College of Science and Technology, Duke Street. 7.15 p.m.

Henlow.—Guidance and Control. L. H. Bedford. Building 62, R.A.F. Technical College. 7.30 p.m.

21st October

Luton.—Fatigue Testing of the Fokker "Friendship." E. J. van Beek. Napier Senior Staff Canteen, Luton Airport. 6.15 p.m.

23rd October

Bristol.—Film Show. Filton House. 6 p.m.

NEWS OF MEMBERS

R. ALLEN (*Associate*) has now completed his service with the Royal Navy and has been appointed a Surveyor to the Air Registration Board.

B. G. AMBROSE (*Associate Fellow*) has completed his Commissioned Service on the active list of the Royal Air Force and has joined the English Electric Company (Guided Weapons Division) as a Senior Designer.

Group Captain A. D. BALMAIN (*Associate Fellow*) has been appointed to command the Central Servicing Development Establishment at R.A.F. Swanton Morley, Norfolk.

Squadron Leader E. BLOTT (*Associate Fellow*) has been appointed Engineer (Electrical) at Headquarters, 2nd Tactical Air Force, B.F.P.O. 40.

K. W. CAMPBELL (*Associate Fellow*) is now employed as a Project Engineer (Hydraulics) with Davy and United Engineering Co. Ltd., Glasgow.

A. V. CLEAVER (*Fellow*), formerly Assistant Chief Engineer, has been appointed Chief Engineer (Rocket Propulsion), Rolls-Royce Ltd. (Aero Division), Derby.

E. W. CORNFORD (*Associate*), formerly Manager of Willis Hole Aviation Ltd., is now Programme Co-ordination Engineer, de Havilland Propellers Ltd., London.

Squadron Leader N. CROWE (*Associate*) has left Normalair Ltd. and is now Technical Sales Representative for The Walter Kidde Co. Ltd., Northolt.

J. R. DIXON (*Associate Fellow*) has left Bristol Aircraft Ltd. and taken a post as Senior Scientific Officer at the Mechanical Engineering Research Laboratory, East Kilbride.

A. F. DUNCAN (*Associate Fellow*) is now employed by the Marquardt Aircraft Company, Van Nuys, California, as an Analysis Engineer in the Performance Analysis Group of the Power Plant Sub-division.

Group Captain W. N. ELWY-JONES (*Associate Fellow*), formerly attached to the Ministry of Supply, P. and E. Establishment at Pendine, has retired from the Royal Air Force and joined the Tube Investment Group of Companies.

S. H. EVANS (*Associate*), formerly Chief Engineer, Field

Aircraft Services, Africa, Ltd., has been appointed Manager of the same Company at their Rhodesian Branch, Salisbury.

H. A. GILBEY (*Associate*), formerly Technical Liaison Engineer with Handley Page Ltd., is now employed in the Public Works Engineers Department at North Beach, Western Australia.

R. C. GODWIN (*Associate Fellow*) has left Marshalls Flying School to take up an appointment as a Service Engineer with de Havilland Propellers Ltd.

Lieut. Commander T. G. F. HARDY (*Associate Fellow*) has been re-appointed from the Staff of the R.N. Engineering College, Manadon, to be Air Engineer Officer of the Sea Vixen Intensive Flying Flight at Yeovilton.

R. C. HOTTON (*Student*), formerly with M.T.C.A. at Bristol, is now an Air Traffic Control Officer with the Ministry of Supply at the Aeroplane and Armament Experimental Establishment, Boscombe Down.

C. N. JAQUES (*Fellow*), formerly Head of Instrument and Photographic Dept., R.A.E., has been appointed Deputy Head of the Ministry of Supply Staff and Director, Guided Weapons and Electronics, at the British Joint Services Mission in Washington.

W. S. LITTLE (*Associate Fellow*), formerly Technical Representative for Shell Oil Co. in New York, has been appointed Aviation Manager, Shell Oil of Canada Ltd.

D. E. MCCULLOCH (*Graduate*) has been released from the Royal Air Force and has resumed his former employment as a Stressman with the Fairey Aviation Co. Ltd., Hayes.

DAVID J. PAYNE (*Graduate*), formerly Technical Engineer with Bristol Aircraft Ltd., is now an Engineer in the Research and Development Group of the Atomic Power Division of the English Electric Co. Ltd., Whetstone.

Group Captain C. RAPLEY (*Associate Fellow*), formerly with H. Button Ltd., Nottingham, has joined the Staff of Hunt Barnard and Co. Ltd., Aylesbury.

R. PARTRIDGE (*Graduate*) has taken up an appointment as a designer at the High Altitude Plant of Rolls-Royce Ltd., Derby.

J. P. RAYNER (*Associate Fellow*) has been transferred from Bristol Aero Engines, Montreal, to Mexico City as Senior Overhaul Engineer with Bristol de Mexico S.A., on a permanent basis.

B. S. SOAN (*Associate Fellow*) has relinguished his post of Assistant Designer with Bristol Aircraft Ltd. and has joined W. S. Atkins and Partners, Consultant Engineers, as Mechanical Engineer to the Research and Development Department.

Rear Admiral S. B. SPANGLER (*Fellow*), formerly of the U.S. Naval Air Development and Material Center, Johnsville, Pennsylvania, has been awarded the Legion of Merit, promoted to the rank of Vice-Admiral and transferred to the retired list. He has joined the Garrett Corporation of Los Angeles as Director of Research and his headquarters are now at the Corporation's laboratories in Phoenix, Arizona.

H. G. SPURR (*Associate Fellow*), formerly at the Royal Aircraft Establishment, is now with the Aircraft Branch of the Ministry of Supply Staff, British Joint Services Mission, Washington.

Flying Officer B. A. STEAD (*Graduate*) was presented by Air Marshal Sir Richard Atcherley, the Commanding Officer of Flying Training Command, with three awards on the occasion of the Passing Out Parade in July. He was awarded the Cup of Honour for the best all-round Student, the Barrett Flying Trophy for greatest proficiency in flying, and the McIntosh Grigor Trophy for the best leadership and officer qualities.

L. H. G. STERNE (*Fellow*) has left the Royal Aircraft Establishment, Bedford, to take up an appointment as Academic Director (Professor) at the Training Center for Experimental Aerodynamics at Rhode-Saint-Genese, Belgium.

JOHN L. THOMPSON (*Associate*), formerly with No. 48 Maintenance Unit, R.A.F., Hawarden, is now an Aircraft Inspector with de Havilland Aircraft Co. Ltd., at Broughton, Chester.

Flight Lieut. J. W. TITTERTON (*Associate Fellow*) has been appointed Engineering/Electrical Officer, No. 4 Joint Services Trials Unit.

F. TODD (*Associate Fellow*), formerly Technician in the Project Department, A. V. Roe and Co. Ltd., Woodford, is now Engineer "A" Special Weapons Division, Canadair Ltd.

H. TONGUE (*Associate*) has recently been granted a Private Pilot's "A" Licence after a course of instruction at White Waltham Aerodrome.

Wing Commander C. V. G. USHER (*Associate Fellow*) has left Cardington and is now the Senior Technical Officer of the Central Servicing Development Establishment, R.A.F., Swanton Morley, Norfolk.

Flight Lieut. A. J. WALDER (*Associate Fellow*), formerly P.A. to the Managing Director, Aero Controls Ltd., is now Works Manager, Alfred A. Corre and Co. Ltd., London.

K. G. WALLACE (*Associate Fellow*), formerly Senior Project Designer at Armstrong Siddeley Motors Ltd., is now working with the Plessey Company Ltd. as Chief Project Designer.

Lieut. A. WORMELL (*Graduate*), formerly at A. and A.E.E., has been appointed Engineer Officer of H.M.S. *Teazer*.

Dr. E. W. C. WILKINS (*Fellow*) has been elected a Fellow of the British Interplanetary Society.

R. E. WILLIAMS (*Graduate*), formerly with British Messier Ltd., is now a Design Draughtsman with de Havilland Propellers Ltd., London.

J. WOLKOVITCH (*Graduate*), formerly with Folland Aircraft Ltd., has been appointed Senior Aerodynamics Design Engineer with Convair at San Diego, California.

W. J. WOODS (*Associate Fellow*) has resigned his post as Chief Stressman, Aviation and Engineering Projects Ltd., to take up a new appointment as Deputy Chief Stressman, de Havilland Propellers Ltd., Conquest House Design Team, London.

D. G. YOUNG (*Graduate*) has left Elliott Brothers (London) Ltd. to join the Engineering Department of the Sperry Gyroscope Company Ltd. (Bracknell Division).

ELECTIONS

The following is a list of new members and transfers of membership of the Society:—

Associate Fellows

Godfrey Lovell Auty (*from Associate*)
Leslie Herbert Bedford
Stanley Arthur Birch (*from Graduate*)
Raymond Lewis Bisplinghoff
Oliver Francis Hawkins Borley (*ex-Student*)
William Brookes (*from Graduate*)
William Kenneth Campe
John Maurice Carey
Michael Albert Catton (*from Associate*)
Richard Alfred Cocke
Denis Eric Cronin
Vincent Ziani de Ferranti
Frederick Haigh Fearnside
Charles Barry Flindt
George Harold Foster
Dennis Victor Hadnutt (*from Graduate*)
Peter David Hall (*from Graduate*)
Eric William Hemingway
Olliver William Humphreys
Leslaw Jerzy Jelonek (*from Graduate*)
Frank Ernest Johnson
James McEwan King (*from Associate*)
John Nevil Kirkby (*from Graduate*)
Clifford James Knights
Karol Kwiatkowski (*from Graduate*)
David Lennox
George Bertram Longden
Donald Ferguson McGregor
Philip de Lacey Markham (*from Associate*)
Michael George Marsh (*from Graduate*)
Eric John Niedermayer
Wallis Parkinson Stack (*from Graduate*)
Derek John Richardson
Julian Gordon de Renzy St. John-Brooks
David Shellard (*from Graduate*)
Cyril Edward Sims
John Edward Talbot (*from Graduate*)
John Graham Towle
Edward Villiers Truefitt
Peter Exton Turner (*from Student*)
Derrick Lewis Wood
John William Herbert Woof (*from Graduate*)
Prithvinath Zutshi

Associates

Richard Joseph Alderson
Douglas Francis Beanland (*from Graduate*)
Roland Herbert Bridgman
Sidney John Hammond
George Ingram
Arthur Samuel Jeffrey
Richard James Merritt
Narayanan Karunakaran Nair
Sidney Nelson Peckham
Frank Ernest Geoffrey Quinn
Leslie Ernest Sidney Tuck
Howard Cecil Tyler

Graduates

William Scott Bannister
Edwin John Dickinson
David John Faddy
Bernard Leslie Robert Fry
Douglas Brownlee Gemmell (*from Student*)
John Geoffrey Hicks (*from Student*)
Roger Martin Wistow Manning (*from Student*)
Kenneth Yuill Marshall
Roderick Andrew Nicol
Raymond Kenneth Oldham
Brian Padgett (*from Student*)
Michael Frank Simpson
Bachalli Raghavendra Rao Vasudeva
August Christiaan Frederik Visman (*from Student*)
John Graham Wakeford (*from Student*)
Peter Edric Wood

Students

Brian William Anderson
Ghassan Raghib Atiyyah
Edward John Charles Brown
Dennis Arthur Bullock
Peter Bernard Edward Engler
Roger John Gabriel
Delme Griffiths
Michael Frank Higginson
Martin George Hole
Denis Raymond Holt
Brian Edward Horne
Kashinath Vishnu Kulkarni
David William Lochtie
Clifford Smith
Peter Rex Smith
Jeffrey John Suter
Ian Morrison Sutherland
Richard Bernard Wade
John Walter Whitehouse
John Stewart Williams

Companion

Lazare A. Florendis

Changes of Address or Appointment

To assist in keeping the records of members correct and up to date the Secretary will be glad if all members will notify him as soon as possible of changes of address. He would also like to know of any change of appointment.

When notifying changes please give the following particulars:—

Name (in block letters) Grade of Membership
New Address (in block letters) Old address

New appointment.—Please given name and address of employer and position held.

Changes of address must be received *before the 15th of the month* in order to be effective for the JOURNAL for the following month.

Journal Binding—New Prices

The greatly increased postal rates, together with other rising costs, unfortunately necessitate revised charges for 1958 for the permanent binding of Journals. The increases are 1s. 6d. on the 1957 volume and 2s. on previous volumes. The new charges are:—

1957 Volume (including packing and postage *in the United Kingdom*) .. .. .. £1 4s. 0d.
Previous Volumes (including packing and postage *in the United Kingdom*) .. .. £1 6s. 0d.

Journals, with a note of the name and address of the sender, should be sent direct to The Lewes Press, Friars Walk, Lewes, Sussex, and the remittance to the Secretary at the Offices of the Society.

Members are asked to be certain that the address to which they want their Journals sent is the same on their letters to the Lewes Press and to the Society.

Self-Binder Cases

It is regretted that the price of Self-Binder cases of the "Easibind" type to hold 12 Journals, which are available from the Offices of the Society, has had to be increased to 13s. 6d. each (*including postage*).

The Journal of the Royal Aeronautical Society

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From Aviation to Astronautics

by

J. E. ALLEN, B.Sc.(Eng.), A.M.I.Mech.E., F.B.I.S., A.F.R.Ae.S.

(Head of Aerodynamics, Projects and Assessment Department, Weapons Research Division, A. V. Roe and Co. Ltd.)

PART I

1. Introduction

Now that major space flight operations are under way in Russia and America there is a natural interest in such topics in this country among those working on aeronautics and guided missiles. At present one can only speculate on how this country will eventually contribute to space flight. As the Duke of Edinburgh has remarked, the difficulty is not due to lack of scientific talent but to the absence of surplus funds[1]. Even so, we can make effective contributions in many ways and we should not lack boldness in seeking out possible ways of doing this. In the meantime, however, there seems to be a place for a review of the present technical situation and prospects for the near future, as seen from the standpoint of one in the British Aircraft Industry. As the majority of interplanetary flights will begin through the Earth's atmosphere and many will finish by way of the same medium, the Royal Aeronautical Society can claim a special and legitimate interest in these matters[2]. Ever since its inception the Society has encouraged the discussion of new technical advances, although its ability to do this has sometimes been hampered by the requirements of security. The same restrictions apply to this paper with the result that I have chosen to draw upon published material throughout, and to review the subject on a broad basis. I hope the work will not be less acceptable on these grounds.

The paper is divided into three parts: the first reviews the present status of aircraft, missiles, rockets and Sputniks; there follow some considerations of possible future design trends in aircraft and missiles in relation to some immediate astronautical problems and it concludes with a discussion of some of the prospects for aeronautical research in very high altitude flight, with suggestions for contributory experiments which might be undertaken in the U.K.

2. History of Aeronautics

It is appropriate to pause and look back to the historical beginnings of aeronautics, for there are many parallels and contrasts between the first flight of the Wrights in an aeroplane in 1903 and the first artificial Earth satellite of 1957. Both represent milestones of human achievement and are successes attributable to man's perseverance and his ability to apply scientific principles. In 1903 the first manned flight aroused little publicity, and such was the state of scientific awareness and international communications that the real stimulus of the event did not become effective until some five years later, during which time many who should have known better denied the truth of the Wrights' claims. The aeroplane, which combined the principles of the windmill and the kite, invented many centuries before, took little from other technologies of its day, apart from the internal combustion engine. The pioneers were individualists and initially the costs were commensurate with their own resources and their private financial supporters. In retrospect these expenses appear trivial, but soon research and construction costs began to feature in government budgets, particularly as the military potentialities gradually became clear. One example[3] of the growth of aviation costs is given in Table I, which clearly shows the effects of war.

The initial incentive for aeronautical development was the satisfaction of the human yearning to discover how to fly, and success was followed by sports flying, military aviation and civil transport. In the succeeding four decades all efforts were concentrated on subsonic aerodynamics and the propeller driven by the internal

TABLE I

Year	*U.S.A.F. cash appropriation millions* $	*Military aircraft costs as fraction of total military costs per cent*	*Military aircraft costs as fraction of total federal expenditure*
1909	0·03	—	—
1914	0·17	—	—
1919	952	—	—
1922	25·6	0·6	0·2
1942	23,049·9	12·7	8·5
1956	15,517	22·5	12·1

Received 12*th May* 1958; *based on a lecture given to the Manchester Branch of the Society on* 22*nd January* 1958.

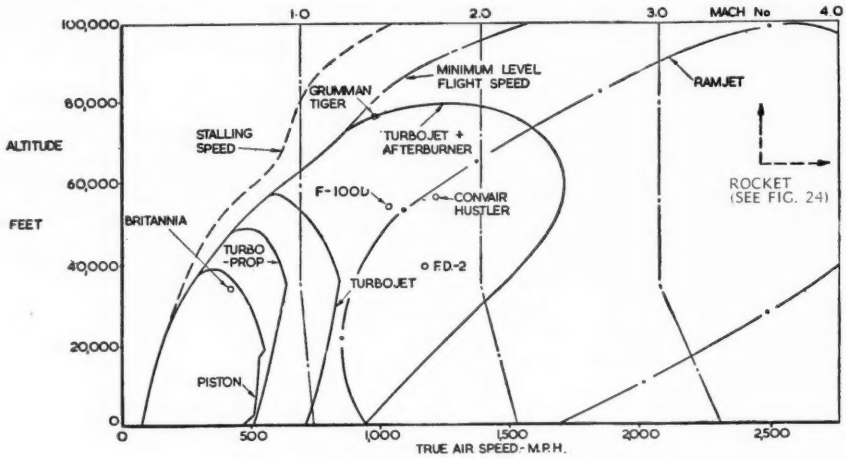

FIGURE 1. Typical aeroplane flight envelopes.

combustion engine was the sole means of propulsion. After 1940, the rate of technical progress speeded up considerably under the impetus of the Second World War and the propeller aeroplane was no longer able to satisfy the increasing demands of speed. Jet propulsion was successfully developed, and so was fully automatic flight in the V.1 and V.2, and the liquid rocket became a dependable source of thrust. The striking advances in performance are illustrated in Fig. 1.

The Sputniks similarly herald a new era of transport as least as different from the aeroplane as the latter was from the train or the motor car. There is a distinct contrast in the publicity given to the Sputniks, for there can scarcely have been a town or village in the world that did not know of them within 24 hours of the first "bleeps." Another significant difference is that the reports were received with universal belief!

From its inception the satellite, unlike the aeroplane, has drawn extensively from the theoretical and practical knowledge established in other fields, such as electronics, guided missiles, and automatic systems, as well as from aeronautics. It is the last fact which makes this transition period so significant to the aeronautical sciences and industries. If no one had prophesied interplanetary flight it would appear that aeronautics, for its own reasons, would have had to develop many similar techniques. In the course of time, however, interplanetary flight will need devices and principles which may not have any direct application on this planet, such as ionic and nuclear plants with radioactive jets[4, 5, 6] (Fig. 2). The use of direct solar energy in one way or another may also be feasible for flights beyond the atmosphere.

3. Flight in 1958

Aviation now plays a major part in the civil and military affairs of the World. All countries are served by the airways and most have Air Forces, some missiles. Vast capital sums are invested in industries, research institutions and facilities and many millions of people work directly or indirectly in designing, building or in using aeroplanes. There seems to be no limit to the great variety of types of aircraft and missiles and before discussing flights beyond the atmosphere it is timely to consider what are the present limits and future possibilities of aircraft.

3.1. HIGH ALTITUDE AIRCRAFT

In the civil transport field, performance, economy and safety have increased steadily over the years. Although piston and turbo-prop engines are expected to play a large part in civil aviation for some time to come[7], the large jet transport aircraft has come to stay in America, Britain, and the U.S.S.R. Rocket propulsion has not yet entered the civil field except for take-off assistance. The nuclear-propelled aeroplane does not look very promising at the moment[8], but a working reactor has already been carried aloft in a converted

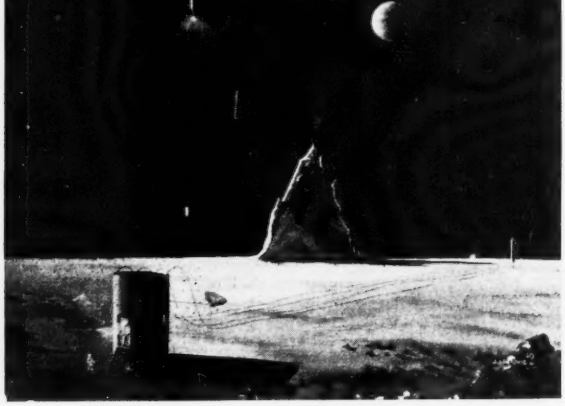

FIGURE 2. Convair nuclear-powered space ship.

FIGURE 3. Convair NB-36H nuclear test aeroplane.

FIGURE 4. Convair B-58 supersonic bomber.

Convair B-36D to investigate radiation effects (Fig. 3). Numerous navigational aids have been perfected and radar and radio communications play vital parts in the control of the congested airways.

Military aircraft are now well through the sound barrier, and a Mach number of 3 should not surprise the pilot of 1958. The Convair B-58, a bomber with a Mach number of 2, flew faster than sound on its first flight (Fig. 4). Altitudes of 126,000 ft. have been reached in experimental aircraft for brief zooming flights[9]. Aviation medicine is, therefore, already dealing with many of the needs of outer space flight, such as pressurisation, oxygen supply, heating, cooling and zero gravity effects. All these and other difficulties will be accentuated in the longer duration flights that can be expected in astronautics[10].

3.2. GUIDED WEAPONS

Never before has a new class of weapon been attended by so much publicity as the guided missile. For centuries new engines of war have been contrived in secret, so that their impact on the unsuspecting enemy might have a more deadly effect. It is difficult to conceal an atomic or hydrogen bomb test, so it is not unknown for foreign observers to be invited to see such experiments. As it is expected that such warheads might be delivered by air, it is now fashionable in some quarters to advertise the missiles as well. When, in addition, the launching sites are placed in such amenable surroundings as Florida or California, it is even more

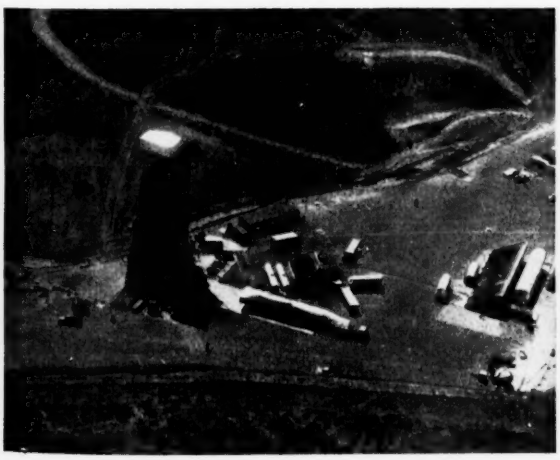

FIGURE 5 (*left*). Atlas arrives for test at the Convair Sycamore Canyon facility.

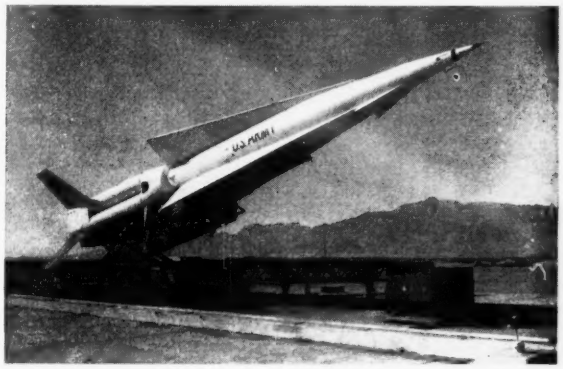

FIGURE 6. Douglas Nike Hercules.

difficult to maintain secrecy (Fig. 5). Missile developments seem to be inextricably caught up as pawns in the political game and the cold war[11]. Because of this, high priority and stupendous financial support has been given to many radically new weapon systems. These developments have, incidentally, made available techniques and equipment well suited to space flight much sooner than might have been expected even 10 years ago.

Interceptor missiles

Air- or ground-launched interceptor missiles have been brought into successful operational use in many countries[12]. Many missile organisations "cut their teeth" on weapons of this type and this experience has created the knowledge of complex electronic control and guidance systems and given an impetus towards higher standards of reliability in electronic components[13]. Of many scores of successful missiles of this class, two examples will suffice: the Douglas Nike Hercules (Fig. 6) which takes an atomic warhead to a range of 40 miles, and the Boeing Bomarc (Fig. 7) which is reported to have a range of the order of 200 miles.

Stand-off bombs

The hitting power and certainty of delivery of the large manned bombers is enhanced by

FIGURE 7 (*right*). Boeing Bomarc.

FIGURE 8. Avro stand-off bomb on a Vulcan.

the introduction of the stand-off bomb which, by being faster and smaller, should evade the interceptor missile. Propelled by rocket, ram-jet or turbo-jet, this class of weapon should be able to attain ranges of hundreds of miles at supersonic speeds[14]. The American Bell Rascal has reached the production and flight test stage; the B-58 can carry a weapon of this class and a British design has also been disclosed (Fig. 8).

Cruise bombardment missiles

Another class of bombardment missile is the long range ground-launched flying bomb, such as the Northrop Snark (Figs. 9 and 10) which flies at a Mach number of 0·9 above 60,000 ft. for 5,000 miles[15] and with high accuracy[16]. Its navigation system supplements an inertial memory with an automatic star tracking unit and the successful operation of this system represents an astonishing advance in instrument and automatic guidance techniques.

Ballistic rockets

Descended from the 200 mile range V.2 of 1944, there are now several ballistic missiles having ranges of 200 to 5,500 miles. The I.C.B.M.'s, such as the Convair Atlas (Fig. 11), weigh about 100 tons at launch and have a range of 5,500 miles in 25 minutes, reaching a peak altitude of about 700 miles[17] in an elliptic path. During the ascent the major requirements are for good propulsion and very high instrumental and guidance precision. For this range the maximum Mach number is about 20. During the descent, the crucial phase is re-entry into the atmosphere, which is approached at a speed of about 20,000 ft./sec. at a moderately steep angle. The friction heating rate is tremendous, and each square foot may be subjected to as much as 3 Megawatts of thermal energy. This creates surface temperatures exceeding 1,000°C., which may last for as long as 15 seconds[18]. It is not a long step from a 5,000 mile ballistic missile to an Earth satellite, a 25 per cent increase in burn out speed being sufficient.

The favoured guidance system for many missiles is inertia navigation, as it does not depend on any radiation and cannot therefore be jammed. Time, gravitation, acceleration and angular velocity are measured with respect to inertial space, and not relative to the Earth, and from these measurements the missile's position relative to its target is computed and steering signals are given to its automatic pilot[19].

3.3. HIGH ALTITUDE ROCKETS

Many different types of experimental rocket have been probing the upper reaches of the atmosphere during the past decade and the great variety of designs demonstrates the advanced state of this branch of technology.

Converted military missiles gave place to specially designed research rockets. Physicists are very interested in the phenomena to be measured at extreme altitudes, but aerodynamicists accept the high altitudes which are the consequence of demanding hypersonic Mach numbers for their test vehicles. A typical multi-stage free-flight rocket from N.A.C.A. sent a test body to a Mach number of 10 and an altitude of 220 miles[20] for aerodynamic and heating experiments. The Lockheed X-17 is a three-stage re-entry test vehicle, reaching 700 miles above the Earth and discharging its nose cone at 15,000 m.p.h. back into the atmosphere[21].

The study of cosmic radiation by means of rockets or satellites was initiated in the U.S.S.R. in 1947 under the direction of the Academy of Sciences[22]. In 1949 a rocket ascent was made leading to formulation of clear requirements for exploration up to 100 km. Special rockets were designed for physics experiments and also for meteorological investigations for the Central Aerological Observatory. These rockets have been in use since 1951.

Dogs were used in experiments between 1951 and 1956. In the first stage of these experiments the animals ascended to 100 km. with a maximum flight speed of 1,170 m./sec. and the maximum acceleration of 6·5*g*.

The work done by the American upper air rockets,

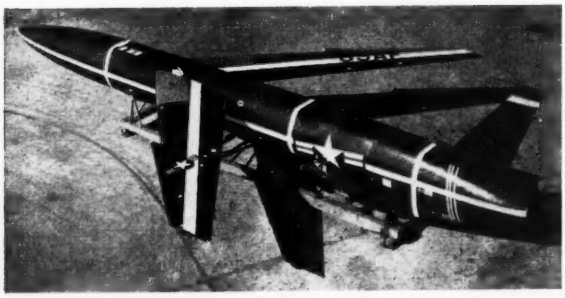

FIGURE 9. Northrop SM-62 Snark.

FIGURE 10. Snark launching.

FIGURE 11. Convair Atlas I.C.B.M. in test stand.

such as Bumper WAC, Aerobee and Viking, is being continued by the participation of the British Skylark and French Veronique in the International Geophysical Year. While these ground-launched rockets explore the first hundred or so miles above the Earth, the altitude can be greatly increased by launching from balloons (such as the Rockair and Rockoon), and the American project "Farside"[23] is now exploring the first few thousand miles. A 3¾ million cubic feet balloon carried a four-stage rocket assembly weighing 1,900 lb. to an altitude of over 100,000 ft. The rockets are fired so that the vehicle passes vertically upwards through the balloon, spending more than an hour beyond the atmosphere. Firings were made in the Pacific in 1957 and it is estimated that an altitude of 4,000 miles was reached. In spite of its remarkably small payload of 3½ lb., this type of vehicle can obtain valuable measurements of atmospheric density, temperature and ionisation. In America, on 22nd November 1957, aluminium pellets were fired by shaped charges from an Aerobee rocket and were estimated to have reached 40,000 ft./sec. and would thus have escaped completely from the Earth's gravitational field[24].

3.4. SATELLITES

The upper air rockets' limitation of only a short time spent at altitude is overcome by the satellite[25]. As Sir Edward Appleton said recently, satellites could be considered as "long playing rockets."

Very full details have been published of the design and operation of the U.S. Navy Vanguard[26, 27, 28]. It has three ballistic stages weighing respectively 17,000 lb., 5,000 lb. and 600 lb. It is ground-launched and is stabilised in flight during the first and second stages by rocket motors whose thrust is controllable in direction. The initial climb is vertical and the vehicle turns over towards the horizontal when the denser layers of the atmosphere have been passed. At a height of 36 miles the speed is 5,500 ft./sec. and at 140 miles altitude it is 13,400 ft./sec. Most acceleration occurs when it is nearly horizontal and beyond the effective atmosphere. It coasts for a few hundred miles before the final rocket puts it into orbit at nearly 26,000 ft./sec. It is spin stabilised before this rocket is fired.

The successful development of this radically new aeronautical device has needed the same dogged perseverance and the ability to solve new scientific problems as were shown by the Wright Brothers. Two new features have appeared: the solution has needed a large industrial and research team and, the availability of substantial financial support from the U.S. Government. This is now exceeding $100,000,000 and it is noteworthy that this was on a civil vote for a purely scientific aim.

The attainment of the high speed of 26,000 ft./sec. with present day liquid and solid rocket propulsion requires three stages, and this is only with uneconomic payload/all-up weight ratios approaching 1,000. The ratio is subject to scale effect and reduces to near 100 for the largest I.C.B.M. type satellite launching vehicle. The precision of final velocity in both speed and direction is clearly critical, but typical tolerances are ± 2 per cent on speed and $\pm 2\frac{1}{2}^\circ$ in direction at burn-out of the last stage[28].

The Russians, also, have evidently solved the two problems of propulsion and guidance with their Sputniks. In a film shown in this country it was suggested that the first stage was winged, but as this was a cartoon it need not be taken to imply the Russians used a different class of design. It is believed to be more probable that the Sputniks were launched Vanguard-fashion from a ground-launched ballistic rocket, the total weight at launch being between 100 and 160 tons.

Essential data on the five artificial satellites so far launched are given in Table II.

The deaths of the first two Sputniks were brought about by air resistance several weeks after they reached their elliptic orbits[29]. Those working on aircraft, who are familiar with altitudes of some tens of thousands of feet and relative air densities of several hundredths, find it hard to form a physical picture of the very tenuous gas at satellite heights. The first satellite was estimated to encounter no more than 1·8 gm. of gas particles in each complete trip round the Earth, of about 27,000 miles. Even at 60 miles altitude the density is a millionth of the sea level value. Most gas resistance occurs over that part of the orbit closest to the Earth. This reduces velocity and substantially lowers the maximum altitude. Eventually the orbit becomes nearly circular and below 100 miles altitude the descent is very rapid[30].

TABLE II

DATA ON ARTIFICIAL EARTH SATELLITES

	Sputnik 1	*Sputnik* 2	*Explorer* 1	*Vanguard TV*-4	*Explorer* 2
Launching date	4th October 1957	3rd November 1957	31st January 1958	17th March 1958	26th March 1958
Weight	184 lb.	1,118 lb.	30·8 lb.	3·25 lb.	31 lb. (approx.)
Size	23 in. dia.	15 to 20 ft. long (final stage attached)	80 in.×6 in. (final stage attached)	6·4 in. dia.	80 in.×6 in. (final stage attached)
Shape	Sphere	Cone-tipped cylinder	Cone-tipped cylinder	Sphere	Cone-tipped cylinder
Angle of orbit to equator	65°	65°	33·5°	34°	34° (approx.)
Orbital period	96·2 min.	103·7 min.	114·95 min.	135 min.	115·7 min.
Perigee	125 miles	170 miles	219 miles	400 miles	125 miles
Apogee	560 miles	1,055 miles	1,587 miles	2,500 miles	1,735 miles
Transmission frequency (Mc./sec.)	40·002 20·005	40·002 20·005	108 108·3	108 108·3	108 108·3
Launching vehicle	Modified I.C.B.M. (?)	Modified I.C.B.M. (?)	Modified Jupiter-C (4 stages)	Vanguard (3 stages)	Modified Jupiter-C (4 stages)

PART II

4. Future Design Trends in Aircraft and Missiles

Aeronautics concerns itself with the carriage of objects from one part of the Earth to another through the atmosphere. For the present purpose interest is concentrated on high altitude and high speed aeronautics. Future design trends will be influenced by the need for improved performance, *i.e.* in altitude, speed and range. These three are inter-related.

Altitude may be needed for its own sake for physical exploration or for the avoidance of attack. Alternatively it may be desired to permit higher speed or increased economy of operations, as in the jet aircraft.

Increased speed shortens the time spent on voyages, reduces the guidance errors of missiles[19], or decreases the chance of their being intercepted. High speed is an essential pre-requisite to range in rocket vehicles of either the ballistic or boost-glide type. Indications are that aeroplanes and missiles will continue to be developed to higher speeds and altitudes. More discussions may still be expected on the relative merits of manned aircraft and missiles, but in the U.S.A. the need for continued development of manned military aircraft is recognised.

Future high speed design is markedly influenced by the two classes of propulsion now available, *i.e.* air-breathing jets and rockets. Turbo-jets and ram-jets have not reached the limit of their development and may attain speeds far exceeding those of today. The N.A.C.A. has expressed a belief[31] in the turbo-jet at a Mach number of 4 (Fig. 12), and a hypersonic ram-jet for $M=6$ has been reported[32]. Whatever advances are made with these systems, however, they are unlikely to be able to operate at altitudes much above 100,000 ft. A good deal of energy is imparted to a vehicle in the first 100,000 ft. of its flight into space and it is probable

TABLE III

CRUISING CONDITIONS AT HIGH ALTITUDE.

Wing loading 50 lb./ft.² Lift coefficient = 0·1

Temperature of undersurface of body at 5° incidence

Surface emissivity = 1·0 Turbulent boundary layer

True air speed (ft./sec.)	*Lift = Weight*			*Lift = Weight − Centrifugal effect*		
	Mach Number	*Altitude* (ft.)	*Mean body equilibrium temperature* (°K)	*Mach Number*	*Altitude* (ft.)	*Mean body equilibrium temperature* (°K)
2,000	2·1	57,500	370	—	Very little difference	Negligible change from column 4
4,000	4·1	86,500	735	4·1	87,500	Negligible change from column 4
8,000	7·9	114,000	935	7·8	116,000	Negligible change from column 4
16,000	13·9	144,000	1,200	13·3	156,000	1,100
20,000	16·7	155,000	1,300	16·3	182,000	1,150

FIGURE 12. N.A.C.A. Mach 4 turbo-jet model.

that air-breathing units will find applications as boosters in some types of interplanetary vehicles because their specific impulse is superior to that of the rocket[33, 34].

Rocket propulsion, being independent of the oxygen content of the upper air, can extend the speed and height of aeroplanes considerably. Table III indicates how Mach number would increase with altitude for a typical supersonic configuration to maintain the same equivalent air speed.

Although rocket propulsion systems could accelerate an aircraft to Mach numbers exceeding 10, the increase of altitude is disappointing and steps have to be taken to cope with the high skin temperatures. The reduction of the wing loading to $12\frac{1}{2}$ lb./ft.^{2} only increases the altitude to 160,000 ft. at a Mach number of 10. 200,000 ft. seems hard to attain in straight and level flight except at speeds near satellite values. Haber has made an analysis of the difficulties of human flight in these regions of the upper atmosphere which he called the aeropause[35].

There have been many proposals for large rocket aeroplanes for both civil and military uses and one example will suffice, namely the plan for a trans-continental passenger carrier put forward by Dornberger in 1955[36] (Fig. 13). This aircraft was designed to fly from London to New York in $1\frac{1}{2}$ hours, and its maximum range was 5,200 nautical miles. The technical requirements seen by Dornberger, *e.g.* controllable rockets, hypersonic configurations, guidance and control, have been tackled vigorously in recent years, but the associated problems of reliability, noise at take-off, and diversion of the glider from a congested terminal airport, will have to be overcome before such devices become attractive for civil use.

There is no doubt, however, that the hypersonic

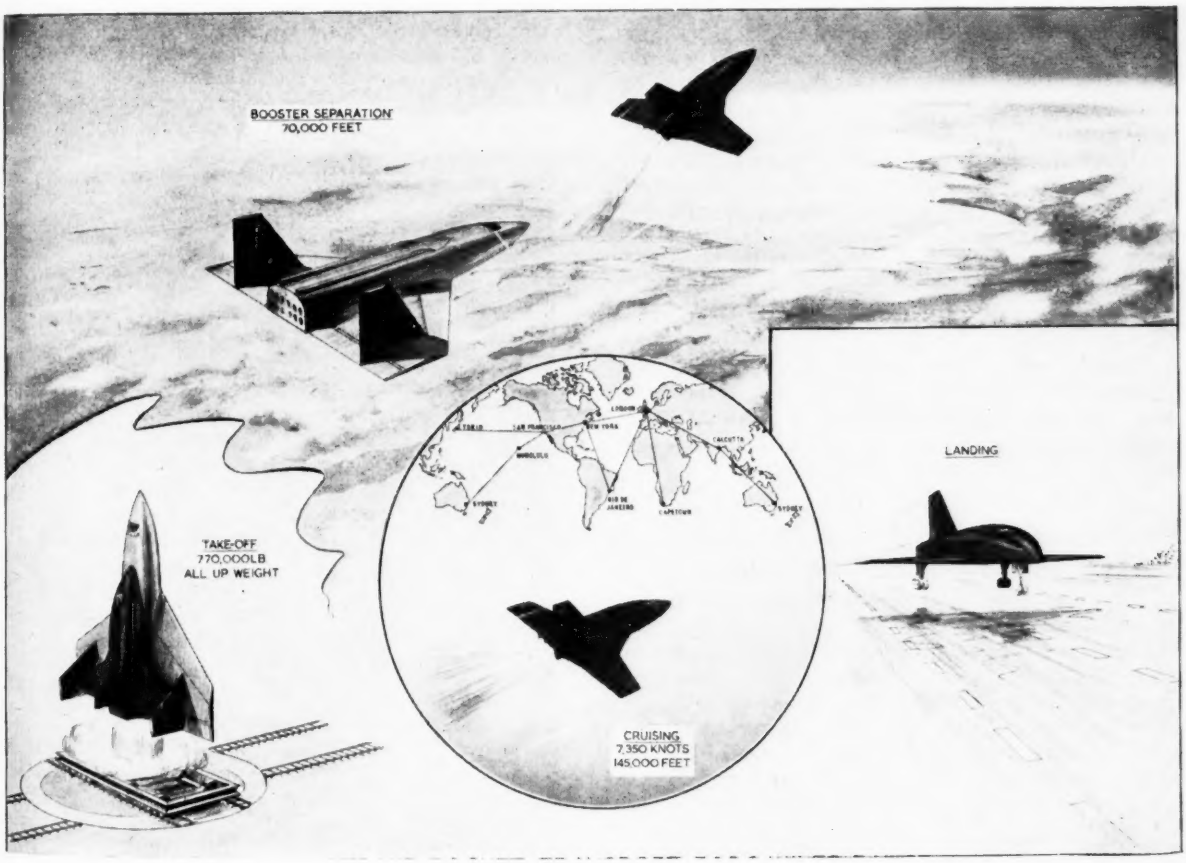

FIGURE 13. Dornberger rocket transport, 5,200 miles range.

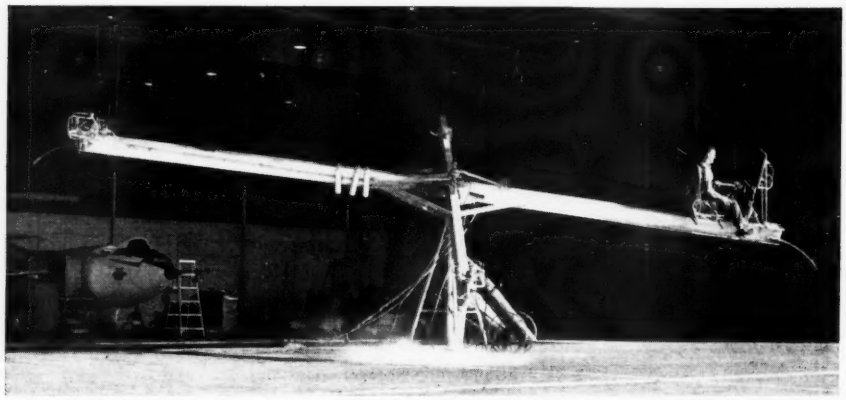

FIGURE 14. N.A.C.A. roll control jet simulator.

rocket aeroplane, missile or interplanetary ferry is at the present time receiving a great deal of attention[37, 38].

The North American X-15, claimed to be the first true hypersonic boost-glide aircraft, is an outcome of this work. It is expected to fly in 1958, probably launched from a Boeing B-52 bomber and, according to press reports, it should exceed a Mach number of 5 and reach altitudes between 200,000 and 800,000 ft., the latter presumably in zooming flight[39]. It is stated to be intended for manned re-entry experiments at altitudes where normal aerodynamic control surfaces are ineffective. Control by adjustable air jets is a means of overcoming this difficulty, and a ground rig for such a system has already been tested by the N.A.C.A.[40] (Fig. 14).

It was once believed that the ground-launched long range ballistic rocket would be the ultimate weapon, but many new systems are now being developed such as the anti-missile-missile, submarine- and air-launched ballistic rockets. There are persistent reports that the Russians are developing a boost-glide bomber, the T-4A, for ranges between 4,000 and 10,000 miles and 120 miles maximum altitude[41], using a 820,000 lb. thrust rocket motor. It is now said to be in the early flight test stages[42].

We may expect, therefore, to see aeroplane and missile performance steadily increasing in the next decade to values approaching those of Earth satellites. During this time many techniques will be shared, and this is considered further in the next section.

5. Science and Technology Common to Aeronautics and Space Flight

It is a commonplace observation that no new device or invention can flourish in the world before its time. The many branches of science and technology associated with aeronautics have received an unprecedented impetus in the past 15 years and the advent of space flight may be regarded as a logical outcome of this work. I shall refer to some aspects briefly, but for reasons given later I have selected aerodynamics and magnetohydrodynamics for closer study.

5.1. ROCKET PROPULSION

This is the only means yet available for escaping the gravitational pull of the Earth. Many billions of pounds have been spent on research and development of the large liquid rocket motors in connection with the V.2 and later with the I.C.B.M. (Fig. 15). The performance of the latter is of the same order as that needed for satellite flights with small payloads.

Table IV gives comparative data of some hypothetical rocket vehicles for terrestrial and extra-terrestrial flights. The estimates are based on plausible parameters[43], but no attempt has been made at optimisation or to relate the values exactly to actual projects. There are two objectives in presenting these figures, firstly to show how launch weight would be reduced by increasing the specific impulse of the propellants from 250 to 350 sec. and secondly, to show how much extra performance is to be gained by replacing a military payload (of a few thousand pounds) by an instrument payload (of about a hundred pounds). A specific impulse of 250 lb. sec./lb. is representative of current practice and 350 lb. sec./lb. could be considered as an ideal target[43a, 43b] for chemical rockets.

FIGURE 15. Atlas rocket test.

TABLE IV

Mission		*200 mile ballistic rocket*	*5,000 mile ballistic rocket*	*Artificial Earth satellite*	*Satellite with some rocket retardation for re-entry*	*Moon landing vehicle*
Characteristic velocity (ft./sec.)		8,000	27,000	33,000	40,000	56,000
Number of stages		1	3	3	4	5
Relative launch weight with military payload (Referred to 5,000 mile ballistic rocket (S.I. = 250) as unity)	S.I. 250	0·04	1·0	3·2	9·6	180
	S.I. 350	0·03	0·3	0·6	1·2	8·4
Characteristic velocity if military payload is replaced by instruments (ft./sec.)	S.I. 250	14,000	34,000	39,000	46,000	60,000
	S.I. 350	18,000	38,000	44,000	51,000	67,000

5.2. GUIDANCE

In automatic inertial and celestial navigation guidance for aircraft and missiles has deserted purely Earth-bound aids to seek the references of inertial space and the stars. The precision of navigation needed for aircraft, missiles and elementary space flight can be attained by comparable instrumental techniques. The accuracy of missile guidance systems has led to revolutionary improvements in component design and manufacturing technique. Linear measurements of micro-inches, angular measurements of seconds of arc, and gyroscopes with a deviation of no more than fractions of degrees per hour are involved[44]. These guidance developments have required the assistance of complex electronic control and computing equipment. The need to cram quarts of electronic brains into pint pots of small interceptor missiles has given a remarkable impetus to miniaturisation techniques such as transistors, digital systems and printed circuits[45]. Recent work indicates that there are still many advances to be made along this road. In the early days of space flight, when performance margins are small, miniaturising equipment will pay dividends in extending physical experiments to greater distances.

5.3. KINEMATICS

In the piston-engined aeroplane, "performance" was defined quite adequately by level speed and rate of climb values, changing with the introduction of jet aircraft to include zooming effect[46]. For very high speed flight one can no longer consider the Earth as flat—a liberty which aeronautics has been able to take for the sake of mathematical simplicity for over 50 years. A new perspective is needed to interpret the flight characteristics of devices capable of near-satellite performance. In this connection, Fig. 16 shows contours of constant total energy representing the sum of potential and kinetic energies. Considerable changes in altitude can be achieved for small losses of speed, but large lateral forces will be required to change direction at these speeds. For ballisic missiles, satellites and interplanetary flights, the trajectories or orbits are solved by the methods of celestial mechanics, established by astronomers many centuries ago. Already maximum altitude is called "apogee" and minimum altitude "perigee."

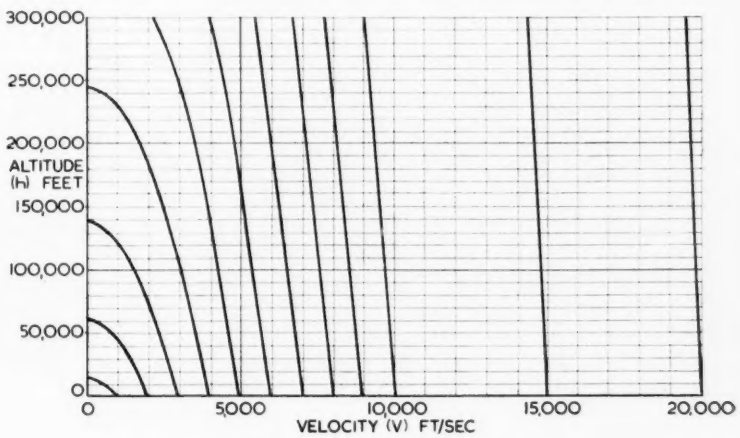

FIGURE 16. Contours of equal $V^2/2g+h$.

5.4. AERODYNAMICS

There has been a major revolution in aerodynamics since 1940. Until then, virtually all aeronautical applications were subsonic and it was obvious that much research was necessary for a better understanding of the theory of lift, drag, propulsion and interference, to attain greater accuracy in design. Since then aerodynamics has progressed through the transonic and supersonic regions to the hypersonic régime. Mach numbers of 23 have been measured in free flight, and 50 and 200[47] (nominal) in the laboratory. We can now truly say that a vast area of aerodynamics has been tentatively explored. The useful limit of Mach number must surely have been reached, for there is no virtue in hurtling through the atmosphere at speeds exceeding satellite velocity. There is a tremendous amount of research still to be done, but the broad physical characteristics of the different régimes have been mapped.

Figure 17 shows the stagnation temperature associated with flight at various Mach numbers through the atmosphere. Temperatures of the magnitude represented will exist behind the strong normal shock waves created by a blunt body (Fig. 18). As the internal energy of the air increases, the higher energy levels such as vibration, dissociation, electronic and ionisation are excited, resulting in much lower temperatures than those obtained from ideal gas theory.

As aerodynamics has extended its scope to these very high Mach numbers, so the research tools and facilities have changed. Wind tunnel designers have manfully kept up with the theoreticians and have produced astonishingly different designs for transonic, supersonic and hypersonic flow up to a Mach number of about 10. Shock tubes and shock tunnels, ballistic ranges and light gas guns extend the range to beyond $M=20$. As the higher Mach numbers introduce intensive aerodynamic heating, the aircraft structural engineer has provided heating laboratories consuming fabulous quantities of controllable electrical power.

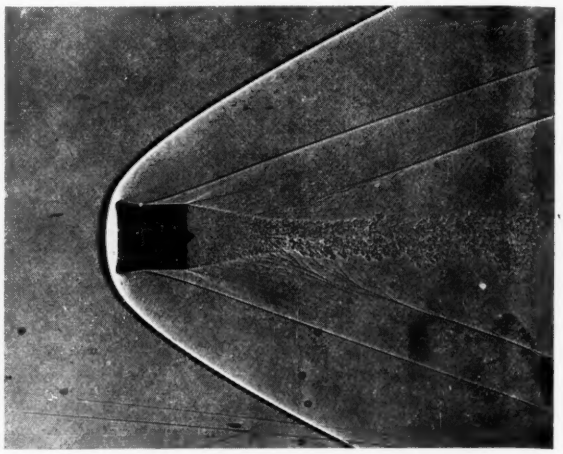

FIGURE 18. Circular cylinder (flat face forward) in the N.A.C.A. free flight tunnel at a Mach number of 2·7.

5.5. MAGNETOHYDRODYNAMICS

In addition to changes associated with the physics of the air at high temperatures, certain chemical transformations take place[48]. At room temperature, air consists of approximately 78 per cent N_2 and 21 per cent O_2 (by volume) in molecular form, with traces of other substances such as water vapour, argon, and so on. At higher temperatures the air contains oxygen and nitrogen in both molecular and atomic forms, and nitric oxide, free electrons and ions may also be present.

At the higher end of the temperature scale associated with high speed flight, air may become sufficiently ionised to act as a conducting plasma which may be influenced by magnetic fields. The study of the motion of an electrically conducting fluid in the presence of a magnetic field is magnetohydrodynamics[49].* The

*Other terms are in use such as magnetogasdynamics, magnetoaerodynamics and hydromagnetics.

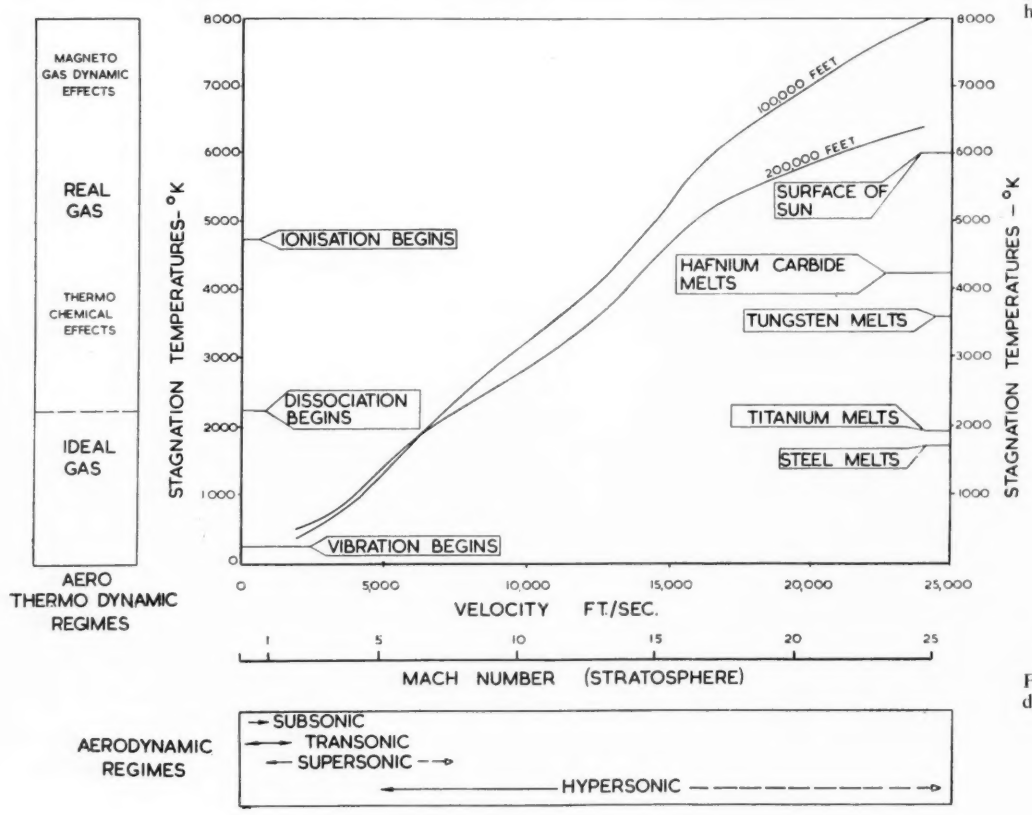

FIGURE 17. Aerodynamic régimes.

equations of magnetohydrodynamics are used to describe such varied phenomena as the production of cosmic rays by solar disturbances[50], sun spot formation[51], flow in the Earth's interior, the stability of spiral nebulae, and the pinch effect used in nuclear fusion experiments[52]. Present aeronautical interest in this subject arises in the plasma jet used either to drive hyperthermal tunnels[53] or to melt high temperature materials[54]. The N.A.C.A. has operated such devices already[55] (Fig. 19). Electrical and magnetic effects are becoming significant in boundary layer flow at Mach numbers near 20[56], and AVCO have made a proposal for utilising electromagnetic drag for re-entry of satellites[57]. A working model of an ion propulsion unit, producing thrust by the acceleration of electrically charged particles in a magnetic field, has been demonstrated by the N.A.C.A. as shown in Fig. 20. Because of the increasing aeronautical interest[58, 59] in this highly involved mathematical subject, I shall describe some fundamental relationships.

The magnetohydrodynamic equations represent the coupling of the hydrodynamic velocity field and the electromagnetic field. The result is a unified system of equations in which the dependent variables (velocity and field strength) fill a six-dimensional space, although the independent variables (x, y, z) are still three-dimensional.

The Euler equations of classical hydrodynamics may be written, in vector notation,

$$\frac{Du}{Dt} = -\frac{1}{\rho}\nabla p$$

where Du/Dt represents differentiation following the motion of the fluid. The Maxwell equations of electromagnetics are

$$\text{div } B = 0$$
$$\text{curl } H = 4\pi\mathbf{j}$$
$$\text{curl } E = -\frac{\partial B}{\partial t}$$

where B represents magnetic induction
H represents magnetic field strength
E represents electric field strength
$\mathbf{j}$ represents current density
and electromagnetic units have been used.

In magnetohydrodynamics it is usual to suppose that the Maxwell displacement currents are small.

Interaction produces an induced electro-motive force

$$E_1 = (u \times B)$$

and a mechanical force (the ponderomotive force)

$$F = (\mathbf{j} \times B) = \frac{1}{4\pi\mu}(\nabla \times B) \times B$$

where μ is the permeability.

From Maxwell's equations for a fluid of negligible electrical resistance, we have

$$\frac{\partial B}{\partial t} = \nabla \times (u \times B)$$

and the equation of motion becomes

$$\frac{Du}{Dt} = P - \frac{1}{\rho}\nabla p + \frac{1}{4\pi\mu\rho}(\nabla \times B) \times B$$

Where P represents non-electromagnetic terms. In order to produce large magnetohydrodynamic effects the magnetic energy and kinetic energy terms represented above should be of the same order of magnitude, *i.e.* $B^2/4\pi\rho\mu u^2 \sim 1$. Ideas new to the aerodynamicist are often involved but at the same time familiar terms like the analogous magnetic Reynolds number ux/η appear where $\eta = 1/4\pi\mu\sigma$ and σ is the electrical conductivity.

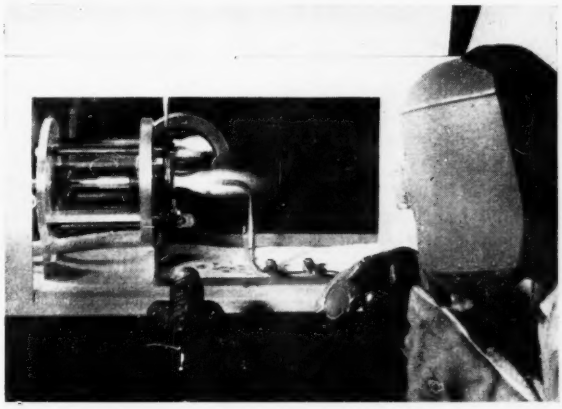

FIGURE 19. N.A.C.A. plasma jet.

The temperature scale illustrated in Fig. 21 shows the magnitude of the temperatures involved in the foregoing discussion, and indicates present and future spheres of particular interest to aeronautics. It should be remembered that the term "temperature" implies the average translational kinetic energy of the particles involved.

In this all too brief review space does not permit reference to such important matters as cybernetics[60], radar, rapid data handling systems, all of which (with many others) are the tools with which the science and technology of astronautics will be constructed.

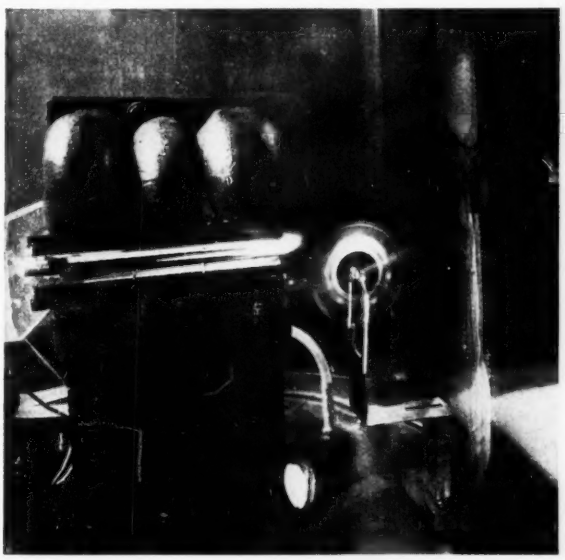

FIGURE 20. N.A.C.A. ion-propulsion model.

PART III

6. Some Immediate Targets in Space Flight*

So much for aviation, aeronautics and the first satellites. Turning now to astronautics, I want to concentrate attention on the more immediate future, say the next 10 years. It is clear that whatever the ultimate objectives of space flight may be[61], a high priority must initially be accorded to experiments measuring the physical, chemical and electrical[62] properties of those regions of space extending a few thousand miles beyond the Earth. The physicist and the astronomer have depended on indirect methods of determining these quantities for several generations, but space flight now gives the opportunity to make direct measurements[63].

The artificial earth satellite is destined to play a key part in this first decade. By extensions of the present techniques, such as better propellants, lighter structures, miniaturised guidance and control gear, and bigger vehicles, larger payload satellites may be expected to go to far greater distances than the first two. The payloads, at first only recording and transmitting instruments, will later be in the form of reconnaissance devices[64, 65] communication vehicles[66], manned observatories and satellites eventually may become stepping stones to space flight further afield. The Russians have officially announced their intention to fire objects to the surface of the moon[67], a task well within the reach of present technical capabilities. They have also spoken of sending transmitting laboratories to the moon's surface. This would require substantial extra development, but it would not be surprising if it were done within the next few years[68]. The first U.S. Lunar probe exploded in flight on 17th August 1958. Other rockets are being prepared[69] to set up moon satellites for physical measurements and transmission of pictures of the lunar surface.

6.1. SATELLITE RECOVERY

The majority of experimental results can be obtained satisfactorily by telerecording, so that inability to recover after re-entry will not seriously hinder this work[70]. It is inevitable that physical experiments will be succeeded by biological investigations, before placing a man in, and recovering him from, a satellite orbit. Safe recovery from a satellite orbit has not yet been achieved and presents technical difficulties which have still to be overcome. Aerodynamics plays a minor role in the launching of a satellite, but it is the dominant factor influencing behaviour during re-entry.

A satellite can be brought down by air drag[71], provided it is not in too large an orbit. Sputnik I showed how unpredictable its final end was, and the occupant of a manned satellite will require more certain knowledge of which week he will return to Earth, whether he will land on a foreign mountainside or in a lonely polar sea, and how hot he will become on the way down.

*The standardisation of units appropriate to flight on the threshold of space is long overdue. A balance should be struck between feet and parsecs, knots and km/sec., and °R and °C, to the greater enlightenment of many of us who are a little confused at the moment!!

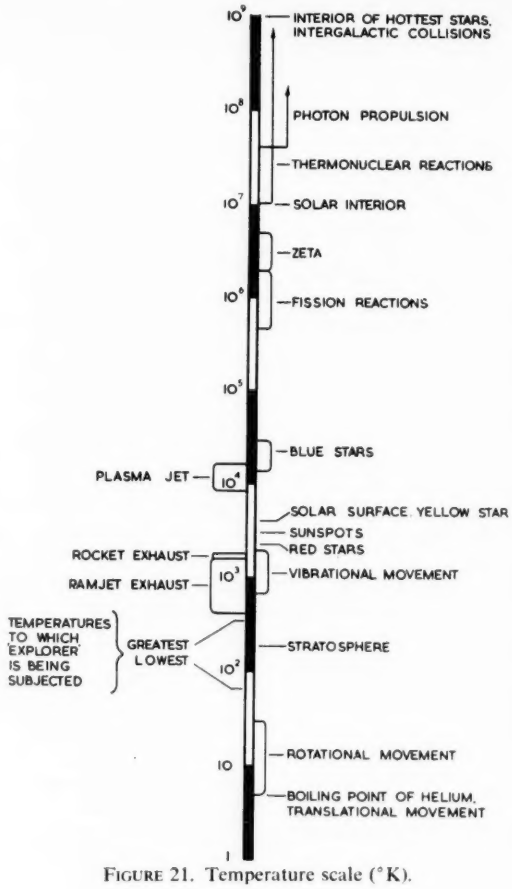

FIGURE 21. Temperature scale (°K).

There are three means of bringing a satellite out of orbit: —

(*a*) By retarding rocket impulse[72]. This may be extravagant in weight.

(*b*) By air resistance. The drag can also be increased by devices such as the variable conical skirt[39] or by steel parachutes[73].

(*c*) By air lift. By this means there could be a greater measure of control over the flight path.

Some theoretical analyses have been made to ascertain if a satellite could reach the earth intact if acted on by aerodynamic drag alone[74, 75]. From these it would appear that the surface temperature can be related reasonably to the parameter (W/C_DA) (where W is the weight, C_D the drag coefficient and A a representative area). This value is close to 10 for the 21 in. diameter Vanguard satellite, and is probably well in excess of 100 for Sputnik II. The larger the value the higher the re-entry temperature. Kemp and Riddell[74] conclude that, for a maximum skin temperature of 1,100°C., (W/C_DA) should be as low as 2, which is hardly practical. Eggers[75] has calculated the re-entry of a large finned hemisphere for which $(W/C_DA)=16$ and which reached a temperature of 1,400°C. A ceramic coating was suggested to withstand this degree of heating. It would be dangerous to generalise from these

brief examples for the data so far available are hardly accurate enough for such calculations. A typical re-entry path is shown in Fig. 22. Although this suggests reasonable temperatures and descent path, it is based on a particular value of (W/C_DA). It may not always be possible to design vehicles for payloads of future interest so that the parameter (W/C_DA) is close to this value.

6.2. RE-ENTRY WITH LIFT

Some project studies have already been made of devices employing lift during re-entry, and the following examples indicate probable trends.

It is shown in Ref. 74 that lift forces applied in the correct sense and magnitude can reduce skin temperatures, but the structures likely to support lift forces may well lead to higher (W/C_DA) values, and there may not be any overall gain in practice. Doubt has been thrown on the ability to develop adequate lift in the free molecule flow conditions obtaining above 200,000 ft.[76].

Reference 74 suggests that to prevent the surface temperature exceeding say 1,400°C. with $(W/C_DA)=20$, the lift-drag ratio should be of the order of 0·6. In the initial stages of re-entry the lift should be positive to maintain a constant heat transfer rate. This lift is not needed for very long and thereafter a ballistic trajectory can be used. Lift/drags of interest in this connection[77] lie between 0·2 and one. Lift-drag ratios of these magnitudes do not seem to be too difficult to achieve with reasonable aerodynamic shapes, even at very high speeds at the altitudes of interest for satellite recovery.

Krafft Ehricke, now of Convair Astronautics, proposed a five-man, three-stage, orbital glider[78] in which the last stage had highly swept wing surfaces. It had a very low wing loading of 11 lb./ft.^{2}, and was estimated to be capable of descending from a satellite orbit and landing on retractable skids. Dr. Ferri[77] has proposed a sled-shaped configuration with highly rounded, swept leading "edges." The payload or crew space is in the lee of the forebody, and re-entry is made in a series of skips. The normal accelerations do not exceed 4g, so that the structure weight penalty is small. Heat generated during a pull-up is radiated away during the subsequent path in free space, and by this means the excessive weight of a heat sink is avoided[74]. Dr. Hilton[80] considered the more severe case of re-entry from an interplanetary path at speeds greater than the orbital value, *e.g.* 7 miles per second. In this case, considerable downward lift is required to curve the trajectory parallel to the Earth's surface. The associated high drag reduces velocity, eventually to below the orbital value. The full brunt of the aerodynamic heating is taken on the flat pressure face which must be cooled or allowed to ablate. A near vacuum is formed on the lee side which reduces heat transfer to the payload.

An enterprising use of lift during re-entry has been proposed by Dr. Eggers of the N.A.C.A.[81] (Fig. 23). A boat-shaped configuration is flown at very high incidence during initial contact with the atmosphere, and in the later stages the best lift-drag ratio is employed. Temperatures (≈850°C) and decelerations (1·7g) are low and the range covered during the descent is about 3,500 miles.

A considerable amount of experimental work must be undertaken before the relative merits of rocket, drag or lift for re-entry from a satellite orbit can be assessed. The best solution may well combine the virtues of all three.

Many have attempted to forecast the pace of

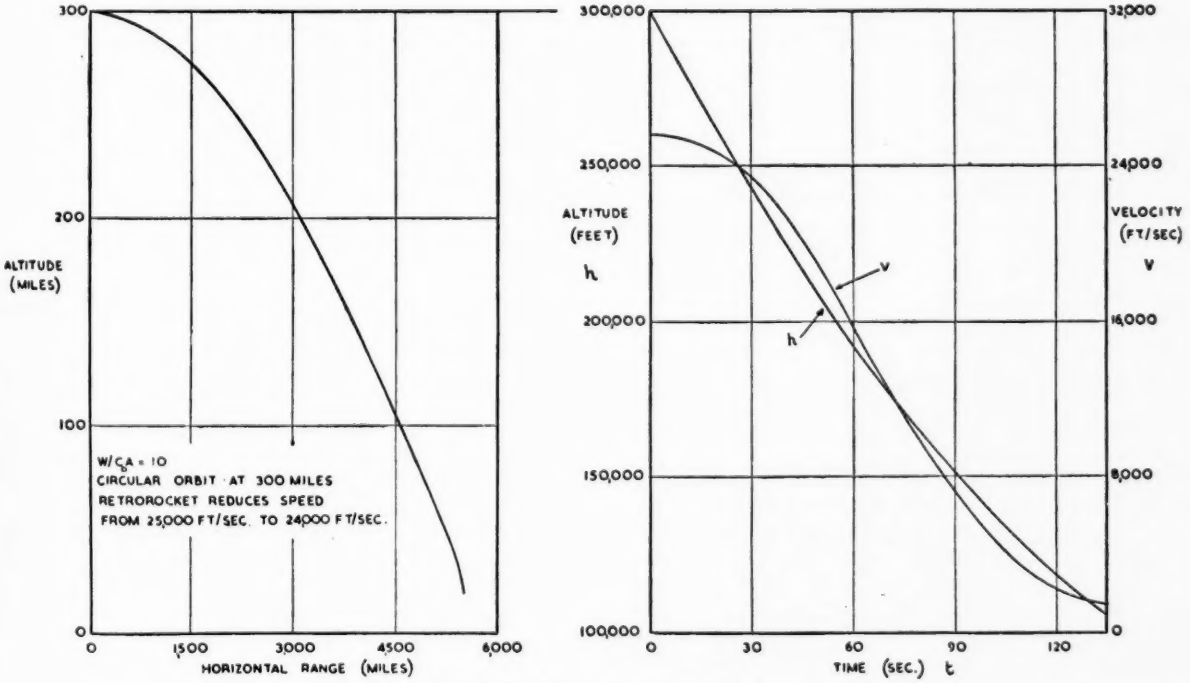

FIGURE 22. Typical satellite re-entry path.

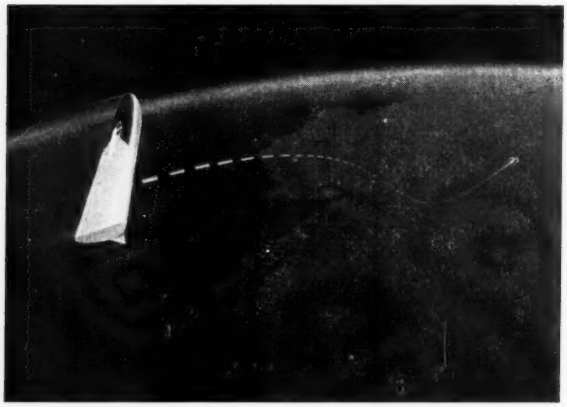

FIGURE 23. Recoverable satellite which may land on water.

astronautical developments in the next 50 years[82, 83, 84]. Others have advised caution in extrapolating beyond the validity of known data[85, 86]; von Kármán does not want to be either too sceptical or too enthusiastic[87]. It is abundantly clear that there is no lack of ideas and proposals for overcoming the known difficulties. Space flight research and development will be extremely costly and availability of financial support will largely set the pace.

7. The Scope of Future Aerodynamic Research at Extreme Altitudes

Aeronautics and space flight are about to embark on a most fascinating period of research[88] at altitudes and speeds which have only been tentatively evaluated so far. We must be prepared for radically new shapes to appear. There are, for example, the ballistic missile with conical or blunt nose re-entering at steep angle, the hypersonic aeroplane[75] with highly swept and rounded leading edges cruising above 100,000 ft., and the hemispherical satellite[89] coming in a low flight path angle to be recovered later by parachute. These three representative configurations have quite different flight characteristics, as shown in Table V.

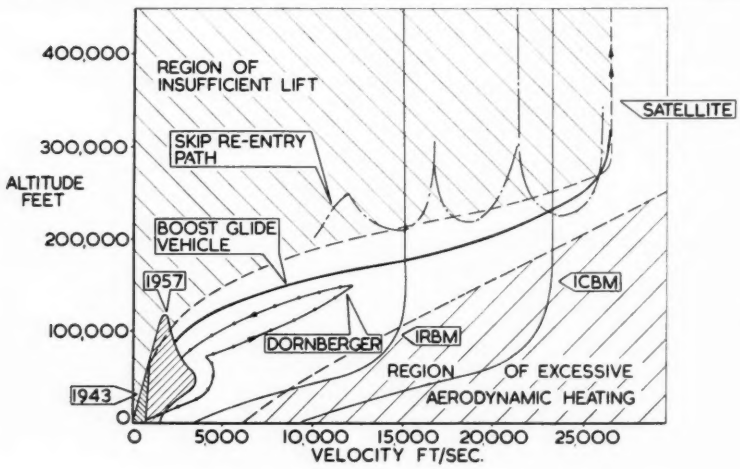

FIGURE 24. Flight corridor.

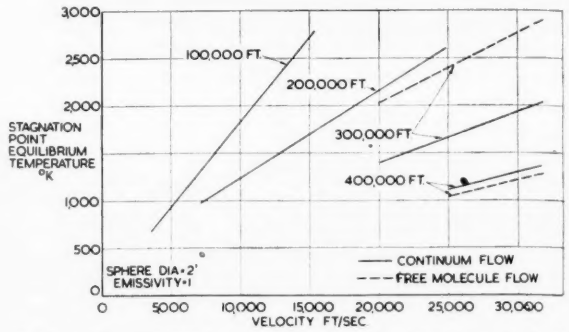

FIGURE 25. Equilibrium temperature in high speed, high altitude flight.

The flight régime in which vehicles of this type operate has been called the corridor of continuous flight into outer space. Its boundaries are not precisely defined, but a typical illustration is given as Fig. 24. This figure also shows the performance attained by aircraft up to 1943 and in 1957, in comparison with the new realm which must soon be opened up.

In the flight corridor, and at greater altitudes traversed by satellites, there are new aerodynamic régimes characterised by the Knudsen number λ/l (where λ is the mean free path of the gas particles and l is a characteristic length of the body). Approximately, when

$$\lambda/l < 0{\cdot}01 \quad \text{we have continuum flow}$$

$$0{\cdot}01 < \lambda/l < \quad 10 \quad \text{slip flow is probable}$$

and when $10 < \lambda/l$ free molecule flow prevails.

The boundaries of these subdivisions are vague and aerodynamic characteristics will depend in particular cases on many phenomena, such as radiation effects, catalytic processes at the body surface[90], and electrical charge effects[91]. A rough guide is provided by possible variations in the equilibrium temperature at the stagnation point of a 1 ft. radius nose ($\varepsilon = 1{\cdot}0$) as in Fig. 25, based on experimental results of Ref. 91(*a*).

Due to the large density increase behind a strong shock wave the region of continuum flow covers a larger range of conditions than is apparent at first sight.

Ionisation of atmospheric gases and radiation effects will influence aerodynamic characteristics and it is clear that the new data will not be acquired easily in these performance regions. Sounding rockets give essentially aerostatic effects, and re-entering satellites have proved to be elusive objects for study during their last few hours of life. Experience and data gained with the re-entry of ballistic missiles will not be wholly relevant to the hypersonic aircraft or the re-entering satellite, for both the angle of dive and the rate of heating are greater, and the ballistic device does not require lift.

TABLE V

FLIGHT PATHS OF UPPER AIR VEHICLES

Class of vehicle	*Altitude* (ft.)	*Mach No.*	*Flight path angle*	*Aero-dynamic heating effects* (kW./ft.²)	*Duration*
I.R.B.M. (1,500 n.m.)	200,000	15	38°	800	15 sec.
I.C.B.M. (5,000 n.m.)	200,000	23	23°	3,000	15 sec.
Hypersonic glider	120,000	5-10	near 0°	5	½-2 hr.
Re-entry from satellite orbit	250,000	20	0-10°	50-1,000	2-5 min.

It is to be expected that several flight techniques will contribute to the exploration of the flight corridor regions and above. As the flight ranges will extend from hundreds to thousands of miles, difficulties will be encountered in measuring trajectory conditions and recovering the objects with due regard to safety. Such flight test vehicles can be ground-launched or air-launched. Air-launching of experimental high altitude vehicles is a well known technique[92, 93]. Fig. 26 shows a recent N.A.C.A. test vehicle[94] which is launched from a Martin B-57A at 45,000 ft. It is a two-stage rocket which reaches a maximum Mach number of 10·5. Air-launching is also suitable for investigating an important class of re-entry flight paths from satellite orbits, which are characterised by flight path angles less than, say, 5°. The potentialities of such a technique, using the Vulcan as the carrying aircraft, were described in the lecture[95] and are illustrated in Figs. 27 and 28. This design study was for a four-stage winged rocket intended primarily for launching an aerodynamic test vehicle at Mach numbers exceeding 15 and at altitudes near 400,000 ft.

No special merit is claimed for the air-launching technique. There are many ways of going about this kind of experiment: some are cheaper than others, and some offer more flexibility.

Flight tests will be supported, and in many cases preceded, by laboratory experiments. For the near future, however, it seems that flight experiments with telemetry will have the primary responsibility of determining the significant aerophysical parameters in near space. After this, ground simulation facilities will no doubt be required and radically new equipment may be expected.

8. Conclusion

I have probably attempted to cover too wide a field in this paper, but my purpose will have been achieved if it has underlined the extent of present day aeronautical activities on the borders of space flight, and shown how many immediate astronautical problems await those engaged in aeronautics. Interest alone, however, is not enough, for I consider that the U.K. should prepare to play its proper part in space flight research and development. It has been said[96] that the participation of Europe in the establishment of space flight is no longer decisively important to mankind as a whole. This feeling, and the costliness of many major space flight objectives, both contribute to a naturally tentative approach from this country. In view of this, we should expect the growth of our research and industrial activity to be somewhat slower than some might wish. Such slowness may not be a bad thing, provided that the time is spent in marshalling our scientific talent and experimental resources in the most effective way.

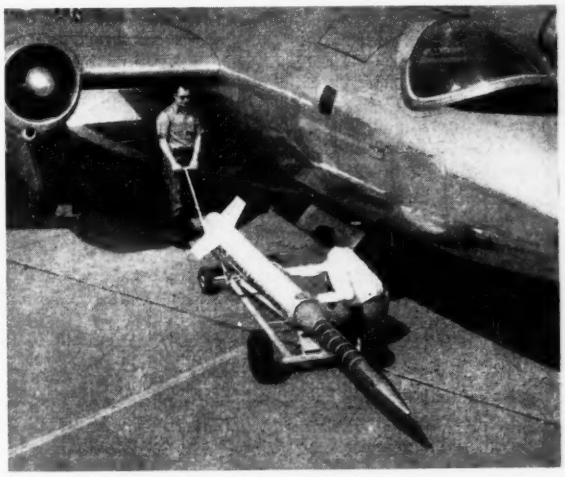

FIGURE 26. N.A.C.A. air-launched rocket.

The manner of approach to space flight in the U.S.A. is well summarised by the President's recent plan for a National Space Agency[97] (Appendix A). The present and future needs of aeronautics and space flight will have many common problems requiring comparable solutions. The aeronautical industries should become aware that space flight operations will contribute to aeronautics, and it would be unwise completely to ignore the incidental results of this work.

For more specific activities, I would recall as an example the part likely to be played by magnetohydrodynamics, a science in which this country may well play a leading part because of its relevance to fusion power techniques.

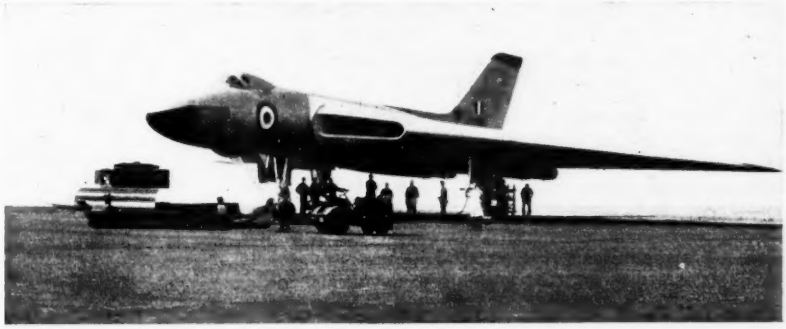

FIGURE 27 Air-launched flight corridor research vehicle (a composite photograph).

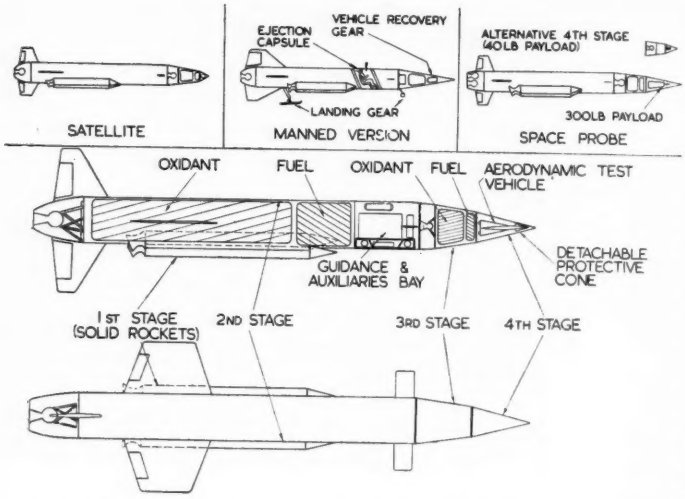

FIGURE 28. Alternative versions of flight corridor research vehicle.

I would once again like to quote from von Kármán who has written[98]: —

> "I think that the 'respectable' scientific and engineering societies should not close their doors to the astronauts or the pages of their journals to papers dealing with the problems of space travel. The present congresses and meetings of the astronautical and interplanetary societies have a relatively high scientific level, especially if we compare them with the activities of certain aeronautical societies only twenty-five years before the first mechanical flight."

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APPENDIX A

PROPOSAL FOR A NATIONAL AERONAUTICS AND SPACE AGENCY IN THE U.S.A.

Brief extracts from President Eisenhower's message to Congress, March 1958.

"In a statement which I issued on 26th March 1958, the Science Advisory Committee has listed four factors which in its judgment give urgency and inevitability to advancement in space technology. These factors are: (1) the compelling urge of man to explore the unknown; (2) the need to assure that full advantage is taken of the military potential of space; (3) the effect on national prestige of accomplishment in space science and exploration; and (4) the opportunities for scientific observation and experimentation which will add to our knowledge of the earth, the solar system, and the universe.

"These factors have such a direct bearing on the future progress as well as on the security of our nation that an imaginative and well-conceived space programme must be given high priority and a sound organisation provided to carry it out. Such a programme and the organisation which I recommend should contribute to (1) the expansion of human knowledge of outer space and the use of space technology for scientific inquiry, (2) the improvement of the usefulness and efficiency of aircraft, (3) the development of vehicles capable of carrying instruments, equipment and living organisms into space, (4) the preservation of the role of the United States as a leader in aeronautical and space science and technology, (5) the making available of discoveries of military value to agencies directly concerned with national security, (6) the promotion of co-operation with other nations in space science and technology, and (7) assuring the most effective utilisation of the scientific and engineering resources of the United States and the avoidance of duplication of facilities and equipment.

"The vigorous programme contemplated will depend not only on adequate legislative authority but also on adequate financial support. I shall shortly submit to the Congress an amendment to the fiscal year 1959 Budget to provide funds that will be needed by the new Agency in its first year of operation."

(It has been estimated that the proposed U.S. Space Agency will cost $206 million in 1959. By 1962 this will have increased to $406 million.)

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AIRCRAFT PROPULSION

A Symposium on Aircraft Propulsion was held at the Derby Branch of the Society on 3rd March 1958 at which the following four papers were presented:—"Some Investigations in the Field of Blade Engineering" by B. D. Blackwell, Engineering Manager, Bristol Aero-Engines Ltd.; "Gas Turbines for Helicopters" by A. W. Morley, Projects Engineer, D. Napier and Son Ltd.; "Fuel Systems for Supersonic Engines" by E. A. Simonis, Systems Engineer, The de Havilland Engine Co. Ltd.; and "Exhaust Nozzles for Supersonic Aircraft" by H. Pearson, Chief Research Engineer, Rolls-Royce Ltd.

Some Investigations in the Field of Blade Engineering

by

B. D. BLACKWELL, M.A., B.Sc.(Eng.), A.F.R.Ae.S.
(Engineering Manager, Bristol Aero-Engines Ltd.)

1. Background of the Subject

The past ten years have seen a wide variety of axial gas turbine aero-engines pass through their development phase into military service. A new phase began when the first axial gas turbine engines to operate to a civil schedule entered service in 1956-7, powering the Britannia and Tu.104 aircraft. The coming years will see an ever-increasing percentage of the world's air traffic being carried by axial gas turbine engines and it may be confidently predicted that in another ten years they will be the rule rather than the exception.

The enormous importance of reliability in civil operation is well known. Possibly less well known is the incredibly rapid build-up of running hours which occurs when an engine is introduced into civil operation. In six months of civil operation these may exceed the cumulative hours in the whole life of a military type, and will outstrip the total manufacturers' bench experience in an even shorter time. With all the achievements in bench development and military service in the past ten years, the axial gas turbine engine is still in the "kindergarten" in relation to civil operation.

Education, in this metaphor, of course, means experimental and theoretical study of the engineering aspects of reliability. While new problems will arise which only years of operation can reveal, we shall do well to make every effort to understand the problems which have already arisen. It is in the nature of aero-engine development that such problems have often been circumvented by *ad hoc* remedies without a full understanding of their cause or the degree of improvement obtained. It is to these problems that it is suggested we should address ourselves in support of the era of axial gas turbine civil aviation which lies ahead.

Taking stock of such problems, those associated with blades emerge high in order of importance. This is not surprising when it is realised that there are some 2,000 fixed and rotating blades in an axial engine and the failure of any one of them can cause it to be shut down in flight and score a black mark for an "unscheduled removal." Thus, for this reason alone, the subject of "Blade Engineering" is felt to be worthy of general attention at this stage of gas turbine development.

However, while reliability in civil operation is a very good reason, there are many others which are equally compelling for the aeronautical engineer. Blades, through their weight and bulk, largely determine the weight of an engine. Blade failures in development cause expensive delays which are particularly embarrassing in these "tight budget" days. Blades amount to about 30 per cent of the price of an axial engine and we must be prepared in the future to compromise intelligently to achieve a reduction of manufacturing cost. This can only be done when we know a lot more than is implied by the "leave-it-alone-it-works" answers often given to the manufacturing side today.

It would be impossible to do justice to this whole field of study in any single paper. The object is only to delineate it by presenting some typical results of an approach to this subject made by the author and his colleagues in recent years. The topics selected have been deliberately chosen to illustrate the scope for further study in this field.

2. The Field of Blade Engineering

Figure 1 shows diagrammatically the field of Blade Engineering as it is considered in this paper. Blade design occupies the central position, the meeting point of the two complementary activities of Design Knowledge and Experimental Investigation. The former

sub-divides into stress, materials and manufacturing methods; the latter into rig testing, engine testing, metallurgical laboratory investigations and manufacturing technique development.

It will be noticed that the performance specification of blading is not included in the field of blade engineering, and the reason for this may be misunderstood. The performance specification of blade angles, solidity, and so on is vitally important—so vitally important that it could be said that no compromise of these requirements is admissible. While this is not strictly true it is the stated object of Blade Engineering in the sense assumed here to take over a performance specification and transform it into a piece of metal meeting the whole spectrum of requirements, including performance. This distinction is believed to be both technically correct, because of the close association of the performance specification with the established field of Engine Performance and Aerodynamics, and practically useful, because it reveals an important area in which little or no co-ordinated work has been done.

Whereas the detailed areas of design study in Fig. 1 are self evident, the experimental facilities are naturally peculiar to one manufacturer and may therefore be obscure to an outsider. The sub-headings under Manufacturing Technique Development, for example, refer to the names of the experimental manufacturing units in Bristol Aero-Engines Limited. Similar facilities will exist in other firms under different names. The four experimental facilities referred to under Rig Testing are:—

(*a*) A single stage rig where the fundamentals of compressor blade vibration may be studied aerodynamically.
(*b*) A turbine rig, using compressed air, where turbine blade vibration may be studied aerodynamically.
(*c*) A variety of electro-mechanical vibration rigs where the mechanics of blade vibration and failure may be studied.
(*d*) A high temperature turbine unit for the study of the failure of turbine blades at operating stresses and temperatures.

Such a wide field can only be covered by a whole range of departments. The work now to be outlined was directed by the author, with the title of Blade Engineer, through a specialist research team calling upon the services of these departments as necessary. The whole-hearted co-operation obtained was an essential and invaluable part of the work.

3. Blade Stress Analysis

3.1. COMPRESSOR BLADE VIBRATION—BLADE FORM FAILURES

The prevalent cause of failure of compressor blades in service is fatigue. Naturally such failures are caused by vibration and while mean steady stresses can contribute to a reduction of life, the focal point of research is a study of the mechanism of blade vibration. It is little exaggeration to say that mechanical design of compressor blades based solely on steady stress considerations is meaningless in these circumstances.

Pearson[1] and Carter[2] provide a useful survey of the existing knowledge of compressor blade vibration. However, while general agreement on the broad character of blade vibration phenomena can be said to exist, serious disagreement exists on the mechanism involved. While this persists, there is little hope of the incorporation of experimental information in the design of blades.

The contribution presented here is the result of a search for a method of describing the vibrations observed in a way which is both practical from the experimental point of view and theoretically stimulating. It is general experience that the discovery of a fruitful pattern into which observations can be fitted is the first step towards a complete theory. Pearson[1] provided the starting point from the theoretical side and our own experimental practice fortunately fitted in.

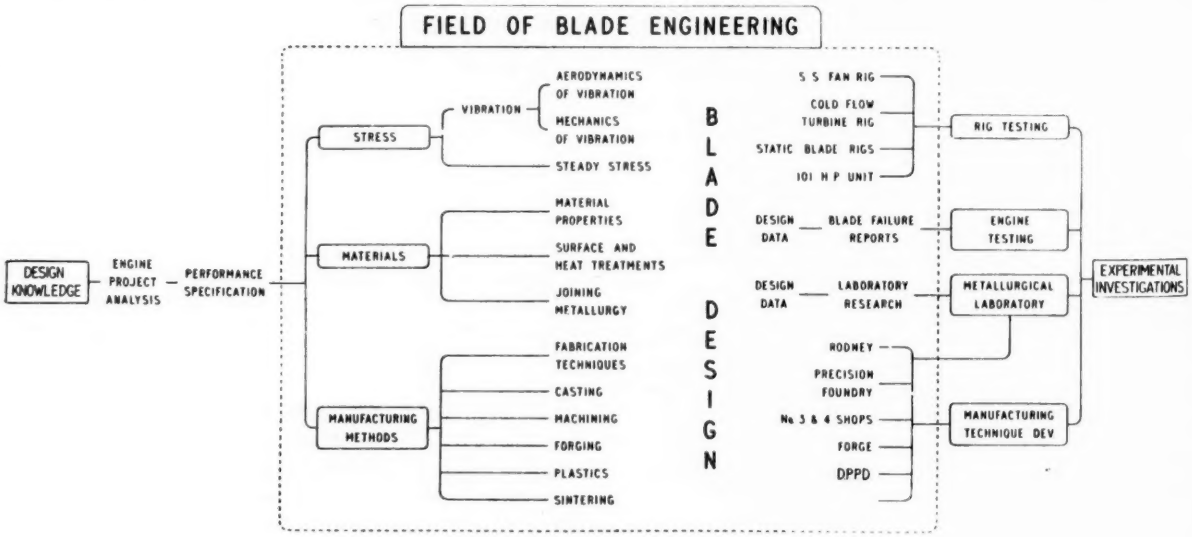


FIGURE 1.

Consider first the problem of failures in the blade form, as opposed to the root fixing:—

Generalising Pearson's approach, two propositions can be stated:

(*a*) The capacity of a cantilever to resist fatigue failure in *any normal mode* can be measured by the parameter $a \times f$ where:
a = semi-tip amplitude in feet (measured at the leading edge point to be specific).
f = natural frequency in c./sec.

(*b*) The capacity of the air to excite vibration is measured by $a \times f$ also; *i.e.* $a \times f$ is a function of aerodynamic parameters only, and independent of the blade material.

Neither of these two propositions is universally true, but each is true in certain circumstances. For example, (*a*) is true for all the flexural modes of an untwisted uniform cantilever on simple beam theory. The expression derived is:

$$a \times f = \frac{1}{2\pi} \cdot \frac{k}{y} \frac{\text{maximum alternating stress}}{\sqrt{(E\rho)}}$$

where k = radius of gyration of root section $= \sqrt{\frac{Ixx}{A}}$

and y = distance from neutral axis to point of maximum stress.

If the section is rectangular, k/y is independent of the shape of the rectangle (length/breadth). If the section is a typical aerofoil, k/y is a constant to an accuracy of ± 15 per cent.

It is of interest to note that the material properties enter as the group $\frac{\text{fatigue stress}}{\sqrt{(E\rho)}}$ as Pearson pointed out.

Proposition (*b*) is true if Pearson's assumptions on the forces operating are valid. These are:

(*a*) All the forces of excitation and damping are aerodynamic.

(*b*) These forces can be calculated by assuming quasi-static conditions (*i.e.* zero frequency parameter).

That these assumptions are adequate to describe certain blade vibrations has been amply proved by Pearson and others. Unfortunately, it is also true that they are inadequate to cover all cases and an example of an exception will be discussed later.

However, let us turn to the severely practical side. All our facts on blade vibration come from measurement and the bulk of this is strain gauge work. Strain gauges in general do not measure the peak alternating strain in any mode and certainly cannot hope to do this in all the modes that emerge in a strain gauge survey on a compressor. This is due to the size of the strain gauge relative to the blade and the varying strain pattern in the blade associated with each mode. To overcome this, we decided many years ago to calibrate the gauge response of each engine blade against leading edge tip amplitude, measured with a microscope on an electrically excited static rig. If blades of this design were then fatigue tested in similar static rigs to failure,

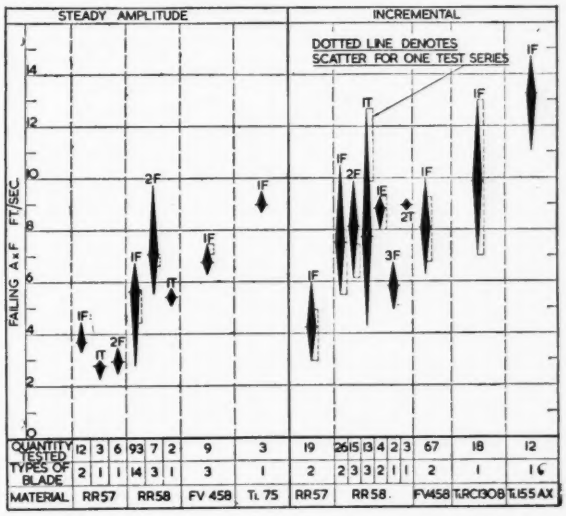


FIGURE 2. Compressor blade fatigue strength for different materials and blade designs.

the meaning of the strain gauge survey could be interpreted in the following manner:

(*a*) The strain gauge signal in each mode was measured in the engine test.

(*b*) This strain gauge signal was converted to tip amplitude, using the static calibration for each mode.

(*c*) The failing tip amplitude was determined in each mode by blade fatigue tests.

(*d*) The seriousness of the vibration was then determined by comparison of the observed and the failing tip amplitudes.

This is a tedious and irritating technique as many of the modes occurring in the engine test were subsequently found to have a negligible amplitude relative to failure after all the calibration testing and analysis work had been done. However, proposition (*a*) reduced this problem enormously. Fig. 2 shows the results of fatigue tests to failure carried out on a wide variety of sizes of blades ($\frac{3}{4}$ in. to 9 in. in length) and materials (aluminium, steel and titanium), plotted in terms of $a \times f$. The left hand side of the diagram shows the results of constant tip amplitude testing and the right hand side of incrementally increasing amplitude tests. The former is the classical laboratory technique in which an *S-N* curve is constructed and is laborious. The latter is a more recent short cut method based on each blade being put through a stepped programme of cycles versus amplitude, the amplitude at which failure occurs being classified the failing amplitude. By this method, each blade contributes a usable result which is not true of the *S-N* method, due to wastage on unbroken specimens, and so forth.

From Fig. 2 it is apparent that a wide variety of blades, materials and modes are brought within a single order of magnitude. Clearly this enables some immediate conclusions to be drawn from an engine strain gauge survey, provided the results are presented in $a \times f$ form. If corrections for blade cross-sectional area

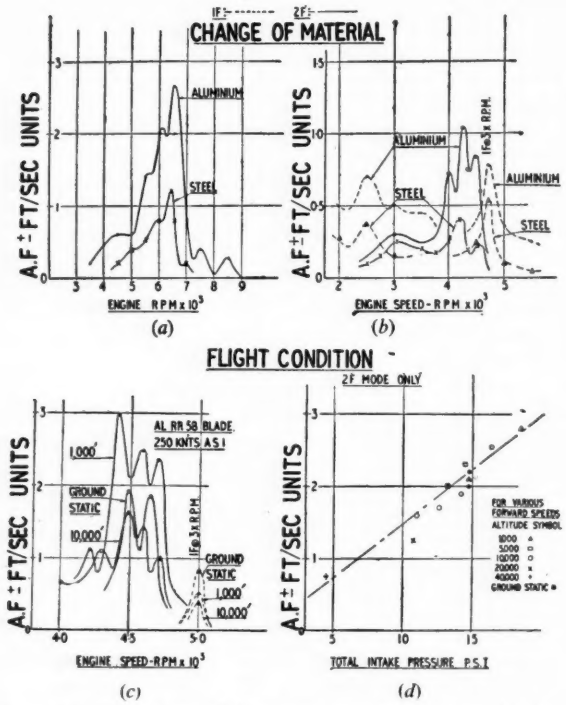


FIGURE 3. Vibrations of first stage compressor rotor blades.

taper and individual section properties are used, the scatter shown can be reduced considerably and the accuracy of interpretation of engine results further improved.

So far only the mechanical aspect of the $a \times f$ approach has been used. The next advantage is illustrated by considering the aerodynamic aspect, and a case in which the Pearson conditions are not obeyed has been chosen as the most searching test. Fig. 3 gives the results obtained in actual engine tests on the bench and in flight, presented in $a \times f$ form. Fig. 3(*a*) shows a troublesome second flexural vibration which caused failure in service. The original blades were in aluminium and it will be observed that the different behaviour of the geometrically similar blade in steel is contrary to the Pearson prediction. However, the virtue of steel is immediately shown not to lie in its strength properties but in some other characteristic that modifies the equations of vibration of the blade. Fig. 3(*b*) shows a similar behaviour in 2*F* on another engine and is also interesting in that the 1*F* resonant vibration also present (3 × engine r.p.m.) shows much less deviation from the Pearson prediction.

Figure 3(*c*) shows the effect of flight on this type of vibration. It will be seen that high intake pressure increases the amplitude and, in fact, Fig. 3(*d*) shows that the peak amplitude is exactly proportional to intake total pressure. Putting all these facts together, they immediately suggest the hypothesis that $a \times f$ is proportional to the ratio air density/material density for this class of vibrations. A possible explanation of this would be that the damping is largely mechanical, although other explanations are possible. The point, however, is that the presentation is immediately useful in stimulating theoretical reasons for the behaviour which could well have been missed if direct strain gauge readings had been presented.

3.2. COMPRESSOR BLADE VIBRATION—ROOT FIXING FAILURES

Structurally it is obvious that a root fixing should always be designed in relation to its blade. This, however, means in relation to the fatigue strength of the blade in its various modes. In other words, we require to know the loads imposed on the root fixing when the blade is vibrating at its failing amplitude in each mode. Fig. 4 shows the results of experiments made to determine these loads.

Standard vibration measurements readily give the frequency and nodal pattern of each mode. The case shown in Fig. 4 is a mode giving a nearly radial nodal pattern which would casually be classified as fundamental torsion. It is, in fact, a complex mode, as will be seen from what follows, and it would be quite incorrect to presume that purely torsional moments are carried through into the root fixing.

The technique used to determine the root fixing forces is to attach short spikes at two extreme positions on each blade section at a number of radial stations up the blade. The blade is excited in the mode required and the movement of these points measured, in magnitude and direction, by means of a travelling microscope. Since the product of amplitude and frequency squared gives the maximum acceleration, and the inertias of each section are known, the maximum forces and couples deriving from each element can be determined. Integration along the length of the blade then gives the root shear forces, bending moments and torque.

Analysis of the movement of the blade in Fig. 4 shows that, in fact, significant transverse and chordwise components of vibration exist in addition to the expected torsional component. The mode is shown to be a compounding of a 1*E* (fundamental edgewise), 2*F* (second transverse flexural) and 1*T* (fundamental

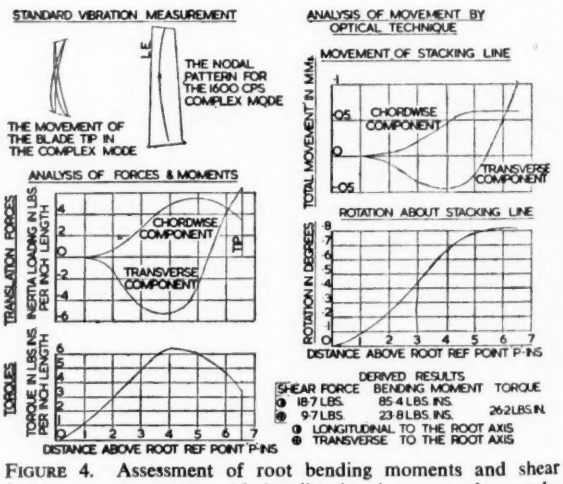

STANDARD VIBRATION MEASUREMENT
THE MOVEMENT OF THE BLADE TIP IN THE COMPLEX MODE
L.E.
THE NODAL PATTERN FOR THE 1600 CPS COMPLEX MODE
ANALYSIS OF MOVEMENT BY OPTICAL TECHNIQUE
MOVEMENT OF STACKING LINE
TOTAL MOVEMENT IN M/Ms
CHORDWISE COMPONENT
TRANSVERSE COMPONENT
ANALYSIS OF FORCES & MOMENTS
TRANSLATION FORCES
INERTIA LOADING IN LBS. PER INCH LENGTH
CHORDWISE COMPONENT
TRANSVERSE COMPONENT
TIP
ROTATION ABOUT STACKING LINE
ROTATION IN DEGREES
DISTANCE ABOVE ROOT REF POINT P-INS
TORQUES
TORQUE IN LBS INS. PER INCH LENGTH
DISTANCE ABOVE ROOT REF POINT P-INS
DERIVED RESULTS
SHEAR FORCE BENDING MOMENT TORQUE
① 18·7 LBS. 85·4 LBS. INS.
② 9·7 LBS. 23·8 LBS. INS. 26·2 LBS. IN.
① LONGITUDINAL TO THE ROOT AXIS
② TRANSVERSE TO THE ROOT AXIS

FIGURE 4. Assessment of root bending moments and shear forces on a compressor blade vibrating in a complex mode.

torsional) vibration. This compounding is, in the main, a consequence of the twist in the blade, with the displacement of the c.g. of the sections from their torsional centre occupying a secondary role (*see* Ref. 3). The resulting forces and moments on the fixing, calculated for unit vibration intensity, *i.e.* $a \times f = 1$ ft./sec., are shown on the table in Fig. 4. These showed that the bending moment in the longitudinal direction was dangerously high for the particular fixing used, although this was quite adequate for the steady design loads. The remedy was to lengthen the fixing to provide a deeper section to resist this moment. Generally speaking, root fixing failures have been traced to lack of appreciation of the high alternating moments in the direction of the root centre line which can be caused by the complex vibration of a twisted blade at modest values of $a \times f$.

3.3. COMPRESSOR BLADE DAMAGE

It has always been obvious that an axial compressor is vulnerable to foreign body damage and experience bears this out. The foreign bodies can take the form of nuts, rivets, pieces of aluminium, stones from the runway, hail and birds. While estimates of the seriousness of the incidence of engine removal due to foreign body damage vary greatly, it is clearly an area of study for the blade engineer.

Damage varies from dents and nicks right up to broken off blades and complete compressor " haircuts." It is the latter type of damage, associated with the larger foreign bodies, that will be discussed here because of the rather unexpected results that have been obtained.

To study the effects of impact on blades, a start was made using an air gun to shoot steel balls at stationary blades. To simulate the type of impact, ball velocities between 100 ft./sec. and 1,000 ft./sec. must be used. The results of the very first experiments, using a half inch diameter ball, are shown in the upper half of Fig. 5. It happened that identical blades in aluminium alloy (RR.57 and RR.58) and in steel (DTD.282 and FV.458) were available in the two cases and the unexpected result was obtained that the permanent damage was the same for steel and aluminium alloy.

In view of this, further fundamental experiments were made with strip specimens mounted as cantilevers. The technique used photography to reveal the total dynamic displacement of the cantilever and the subsequent vibration, as shown in the lower part of Fig. 5. From a series of experiments of this type, it was found

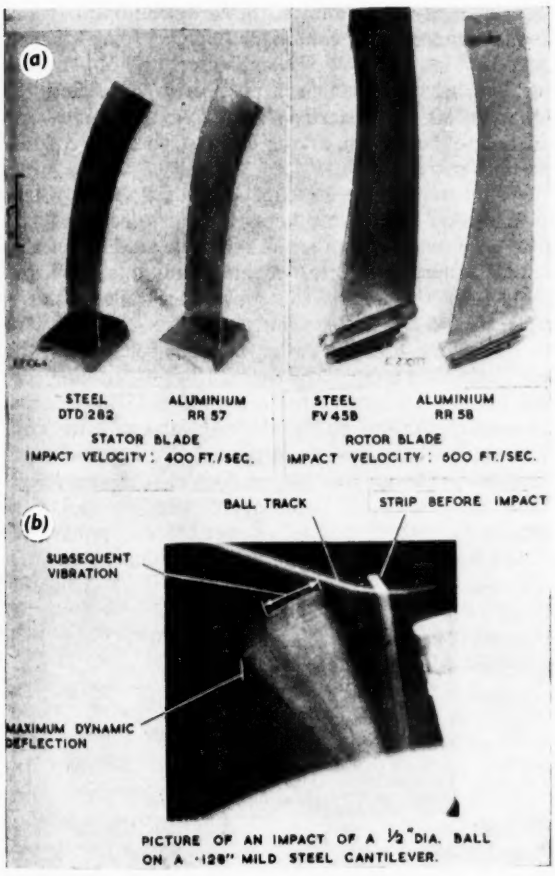
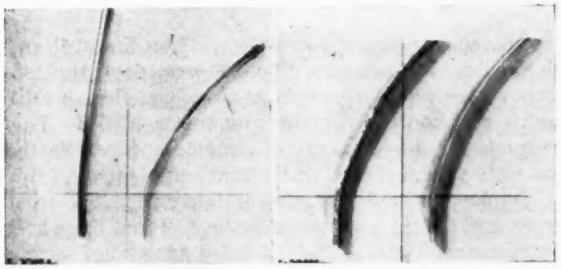


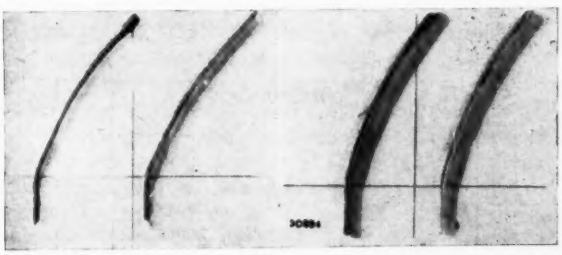
FIGURE 5(*a*). Instances of identical impact on aluminium alloy and steel blades producing comparable damage.
(*b*). The technique of dynamic displacement measurement.



(A)

(1) STEEL (MILD STEEL) BALL VELOCITY 204 ft./sec. Comparison shows different impact damage with:— 0·064 in. thickness, ¼ in. dia. ball 0·128 in. thickness, ¼ in. dia. ball

(2) ALUMINIUM (DTD 182) BALL VELOCITY 204 ft./sec. Comparison shows same impact damage with:— 0·064 in. thickness, ¼ in. dia. ball 0·128 in. thickness, ¼ in. dia. ball



(B)

(1) STEEL (MILD STEEL) BALL VELOCITY 204 ft./sec. Comparison shows same impact damage with:— 0·064 in. thickness, ¼ in. dia. ball 0·128 in. thickness, 1 in. dia. ball

(2) ALUMINIUM (DTD 182) ¼ in. DIA. BALL Comparison shows same impact damage with:— 0·064 in. thickness, 165 ft./sec. ball velocity 0·128 in. thickness, 230 ft./sec. ball velocity

FIGURE 6. Strip cantilever impact results.

that aluminium and steel followed laws of a different nature, thus:

$$\text{Aluminium:}\quad \frac{\Delta}{l} \propto \frac{MV^2}{m}$$

$$\text{Steel:}\quad \frac{\Delta}{l} \propto \frac{MV^2}{mt^2}$$

where: Δ = permanent deflection at tip.
l = cantilever length.
M = mass of the ball.
V = velocity of the ball.
m = mass/unit length of the cantilever.
t = thickness of the rectangular section.

The unexpected consequences of these different laws are shown in Fig. 6. The law for aluminium, for example, predicts that the same permanent deflection will occur if both the thickness of the strip and the mass of the ball are doubled and the effect of this is illustrated in the top right hand picture of Fig. 6. However, if this experiment is repeated for steel, a totally different result is obtained, as shown in the adjacent upper picture. In the lower left hand picture it is shown that it is necessary to increase the ball mass by 2^3 to produce the same permanent displacement in a mild steel strip of twice the thickness. In the lower right hand picture, the effect on aluminium of doubling the thickness and doubling the ball velocity squared is shown to leave the same permanent displacement, as predicted in the formula.

The effect of these two different laws for steel and aluminium is to produce different permanent displacement curves which cross over at a thickness/length ratio which commonly occurs in compressor blades. This accounts for the similarity of damage apparent in the two tests on actual steel and aluminium blades.

It may be said that this is interesting, but what bearing has it on compressor design? From the whole programme of work, which included analysis of service damage, the following points emerged:

(*a*) The most likely cause of major damage is the dynamic deflection of the blade which is a maximum with tip impacts. The damage then follows due to impact between the stationary and rotating rows, tearing off a blade and precipitating a chain reaction of blade failures.

(*b*) The laws governing the dynamic deflection for different materials are different in a manner for which no explanation is yet available.

(*c*) It requires very large masses to shear off a blade due solely to impact without the assistance of impact with neighbouring rows of blades.

(*d*) It follows from the above that adequate axial clearance between rows can reduce damage effects. Adequate may be loosely defined as such that the dynamic deflection (which has an axial component due to the stagger of the blade) causes a blade to strike its neighbour in the same row before it can strike the next row.

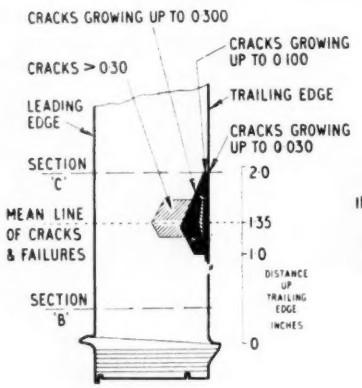
3.4. TURBINE BLADE-FORM FAILURES

The lower temperature stages of the turbine are in a very similar position to compressor blades, and when they fail they invariably exhibit signs of fatigue. Much of what has been said about compressor blade vibration problems then reads directly across.

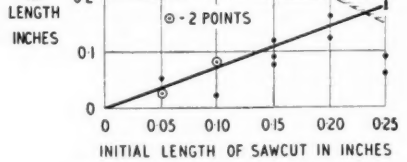
However, the first stage blades in a turbine operate in a creep régime where their hours of life are numbered even if no vibratory stresses are present. Clearly steady stress analysis has a more important rôle to play under these conditions. This is not to imply that vibration and cyclic thermal stress do not play their part. It has, however, been possible to correlate blade failure effects with steady stress *alone* and this is considered to be an important finding, possibly contrary to some current beliefs.

The basis of this investigation was a series of tests to failure on the 101 h.p. unit referred to earlier. This is to all intents an engine in which a bladed wheel is run to failure at typical engine conditions. When a blade fails, the unit is shut down, the failed blade is replaced and the test continued.

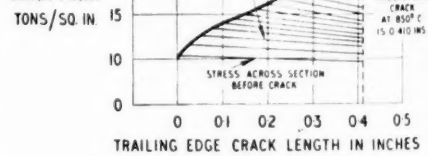
In the first test, the design of blade selected for this work exhibited a fracture starting from the trailing edge and running straight across a section 1·35 in. up from the root platform. This point coincided with the position which would have been predicted using the known combustion temperature profile and the centrifugal stress curve *only* (*i.e.* ignoring all bending moments). Subsequent work has confirmed, with remarkable accuracy, this point of failure. Continuing the test



Distribution and growth of trailing edge cracks



Growth of sawcuts to simulate cracks.



Stress versus crack length.

FIGURE 7. Crack propagation.

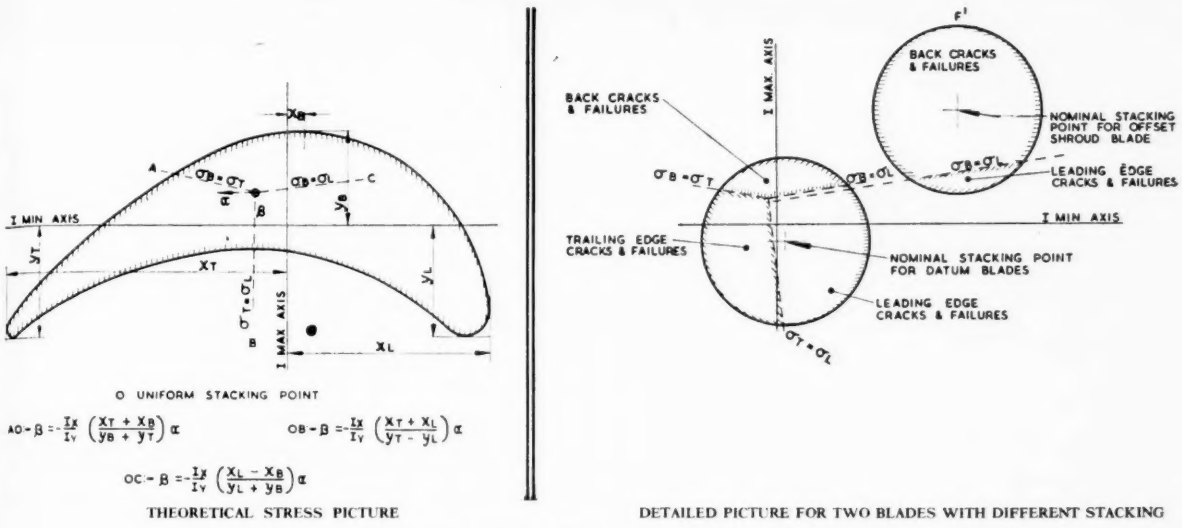

I MIN AXIS
I MAX AXIS
O UNIFORM STACKING POINT
THEORETICAL STRESS PICTURE
BACK CRACKS & FAILURES
NOMINAL STACKING POINT FOR OFFSET SHROUD BLADE
LEADING EDGE CRACKS & FAILURES
TRAILING EDGE CRACKS & FAILURES
NOMINAL STACKING POINT FOR DATUM BLADES
DETAILED PICTURE FOR TWO BLADES WITH DIFFERENT STACKING

FIGURE 8. Influence of stacking on crack propagation.

programme by replacing the damaged blade and recording the cracks in the unfailed blades, the pattern of crack development shown in the far left hand corner of Fig. 7 was found. Small cracks were distributed over a wide area around the final failing section, but only those in the section 1·35 in. up propagated extensively and culminated in another failure, identical in position to the first.

Continuing this test programme, replacing failed blades with new ones, the longer life blades started cracking at the leading edge, and finally a failure (still at 1·35 in.) showing a leading edge nucleus occurred. After still further tests replacing failures as they occurred, the " Methuselahs " of the original set started cracking from the back. At this stage the testing was discontinued, but it seemed reasonably certain that a failure from the back of the oldest blades would have been obtained.

A subsidiary area of research was now started. Sawcuts of various depths were put at the failing section in a new set of blades, and the result shown in Fig. 7 obtained. With some scatter the increase in crack length in a given time was found proportional to sawcut depth, *i.e.* the rate of crack propagation was proportional to crack length. The calculated stress picture appropriate to this saw-cutting is also shown in Fig. 7 and it will be seen that this result is broadly explained by the steeply rising stress at the head of the crack as it penetrates.

Now consider the simple stress analysis in common use for turbine blade work. The applied loads on any given section through a blade are resolved into:

(*a*) A centrifugal force operating at the centre of area of the section due to the centrifugal action of that part of the blade outboard of the section being studied.

(*b*) A centrifugal bending moment due to the fact that the line of action of this force does not, in fact, pass through the centre of area.

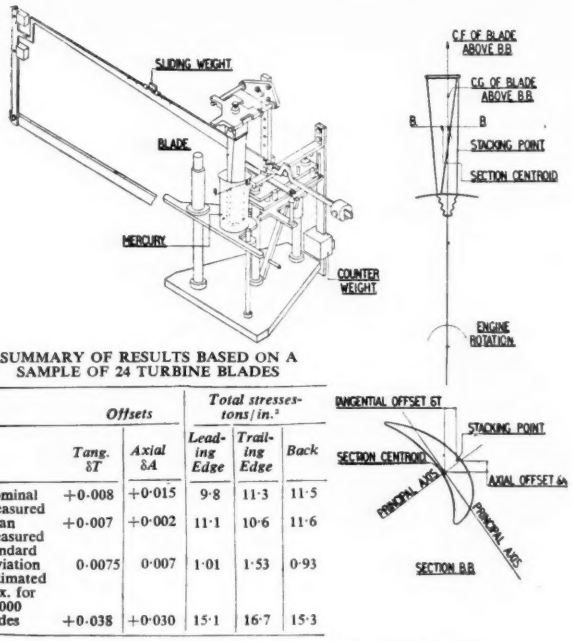
(*c*) A gas bending moment due to the externally applied aerodynamic loads.

Other forces are usually negligible and the net result is that there are three points on the blade, loosely termed leading edge, trailing edge and back, where the stresses may be critical. If only moments about one principal axis of the section were applied, these points would be located precisely on the periphery of the blade farthest from this axis. However, this is not true, and so the three critical points are not simply defined and the above description only specifies the broad area in which they fall.

It was decided to study the effect of the point of application of the centrifugal force to explain the position of initial cracking. This subject is called "stacking" because the point of application can be influenced by the way the sections are "stacked" one on top of the other. For example, if the blade were to be leaned over, it is clear that a significant re-distribution of stress on any section would occur, although the centrifugal stress would be unchanged to a first order.

Figure 8 shows the theoretical effect of the choice of stacking point on the location of the maximum theoretical stress. Given the gas bending loads, a diagram can be constructed consisting of the three zones shown. These zones meet at a point called the uniform stacking point, such that if the centrifugal force passes through this point then the section is equally stressed at all points and this stress is the simple centrifugal stress. Away from this point, the maximum stress occurs in the leading edge, trailing edge or back depending on the zone in which the stacking point falls.

The datum blades used in these tests were stacked as shown in Fig. 8. If a circle is drawn with the nominal stacking point as centre to represent the effect of tolerances in manufacture, it embraces all three zones and immediately explains why failure may be initiated from leading edge, trailing edge or back. Furthermore, it can be shown by calculation of the various stresses so



SUMMARY OF RESULTS BASED ON A SAMPLE OF 24 TURBINE BLADES

	Offsets		*Total stresses-tons/in.²*		
	Tang. δT	*Axial δA*	*Leading Edge*	*Trailing Edge*	*Back*
Nominal	+0·008	+0·015	9·8	11·3	11·5
Measured mean	+0·007	+0·002	11·1	10·6	11·6
Measured standard deviation	0·0075	0·007	1·01	1·53	0·93
Estimated max. for 10,000 blades	+0·038	+0·030	15·1	16·7	15·3

FIGURE 9. Turbine blade stacking measurement.

produced, that the order in which failure occurred is in accord with the calculated stresses for each nucleus location.

To test this hypothesis further a new set of datum blades were made with the blade tip shroud modified by adding material over the top on one side. This slightly increased the C.F. stress but displaced the stacking point significantly, as shown on Fig. 8. The life to first failure was significantly larger than the original datum set, and the failure occurred from the back, as predicted by this theory.

The true statistical scatter in individual stacking points was not known directly at this time. Normal inspection methods involve the integration of a lot of discrete dimensional information to obtain the stacking point location of a given blade. This is laborious and inaccurate. The method shown in Fig. 9 was therefore adopted. It relies on the hydrostatic principle for obtaining the centre of gravity of an irregular shape. By weighing in air and mercury in two known positions, enough equations are available to solve for the position of the stacking point.

The results of the application of this method are also shown in Fig. 9. First the position desired by the designer and that obtained in manufacture are shown, and it will be seen that a major discrepancy exists in the axial direction. The scatter, as measured by the sample of 24 blades, is shown as a standard deviation of the axial and tangential readings. It is of interest to compute the consequences of this scatter, assuming a Gaussian distribution, on the highest stresses occurring in the three critical areas for 10,000 blades. This number coresponds to 100 engines with 100 blades per wheel in service. It will be seen from Fig. 9 that the worst blades can have an increase in local stress of some 5 tons/in.² above the nominal value. The implications of this in terms of scatter in life to cracking are very serious, and isolated cases of cracks found at overhaul are readily explained.

To conclude, it appears that by introducing the concept of stacking point, and its scatter due to tolerance, the following observed facts are explained:

(*a*) The increase in life to be obtained by careful selection of the nominal stacking point which should be based on an allowance for variations due to tolerances.

(*b*) The scatter in life, and different forms of failure, of supposedly identical blades.

(*c*) The preponderance of leading edge and trailing edge failures, often put forward as proof of the importance of "thermal shock."

It seems remarkable that the very simple stress concepts can get so far towards an explanation of actual blade behaviour. It is not, of course, intended to imply that other mechanisms of failure do not operate and will not also prove to be of great importance.

4. Blade Materials

4.1. THE DERIVATION OF DESIGN DATA FROM LABORATORY MECHANICAL TESTS

It is generally known that there is some difficulty in correlating laboratory tests on materials as between one laboratory and another. However, attention will be drawn here to the equally important problem of deriving design data from laboratory data. It is invariably true that laboratory data is of no use for design purposes in its raw state and must be interpreted before it can answer the design and development engineer's questions. Fig. 10 illustrates the problems of interpretation for a hypothetical blade material. These problems may be explained as follows:—

(*a*) *Creep Data*

Interpolation is essential because laboratory data gives discrete points and never covers the whole field required by the designer. However, if this data is plotted in the manner suggested by Larson and Miller[4], a master curve can be constructed based on actual test values which permits the derivation of stress versus temperature curves for any arbitrarily chosen number of hours to failure. Thus the curves of Fig. 10(2) can be derived for the design life required. These, however, have to be amended because it seems proper that the designer should take account of scatter which is implicit in the specification minimum properties used for acceptance of material in production. This means a choice of a factor by which the laboratory properties are reduced, in this case 0·15, based on a special investigation of the case under consideration. This is shown in Fig. 10(3).

(*b*) *Fatigue Data*

Figure 10(4) shows a fatigue data in the form normally presented by laboratory tests. This data is usually of the reverse bend type with zero mean stresses, but in some instances the designer may have some data available with superimposed steady tensile stresses. If

the latter is not available the usual straight line assumption joining the extreme points on the two axes of the Goodman diagram is made. The first difficulty arises when this data has to be extrapolated. The practical limit to a 2,000 cycle per minute laboratory test is about 1,000 hours, but this represents only 100 hours or less at blade frequencies. Fig. 10(5) shows the simple straight line expedient which has to be adopted because of this deficiency in the data.

The next dilemma is concerned with the interconnection of creep and fatigue at the high temperature end. This may occur above 200°C for aluminium alloys or 600 – 700°C for nickel-based alloys. By plotting the fatigue data based on number of cycles over the low temperature region and the creep rupture curves at the high temperature, some faired curve between these two extremes can be guessed. Figs. 10(7) and 10(8) show the conversion of this fatigue data into $a \times f$ terms, stemming from the arguments presented earlier in this paper. This is done by using the laboratory data to obtain the form of the curves and using actual blade fatigue test correlation factors to convert to a practical $a \times f$ figure for blades.

(c) *Moduli*

Figure 10(9) shows the remaining properties that are required for calculation in the Design Office plotted against temperature and completes the design data sheets.

In many respects the processes described are unsatisfactory and are put forward to show that there is no grounds for complacency in this matter. It is an area in which only the closest co-operation between the engineer and the materials test laboratory can overcome the difficulties and replace assumption by fact.

4.2. THE SPECIFICATION OF HEAT-TREATMENT

The next area for discussion after selection of the material is its heat-treatment. This problem again lies at the front between metallurgical and mechanical engineering, since it is possible for the metallurgist to

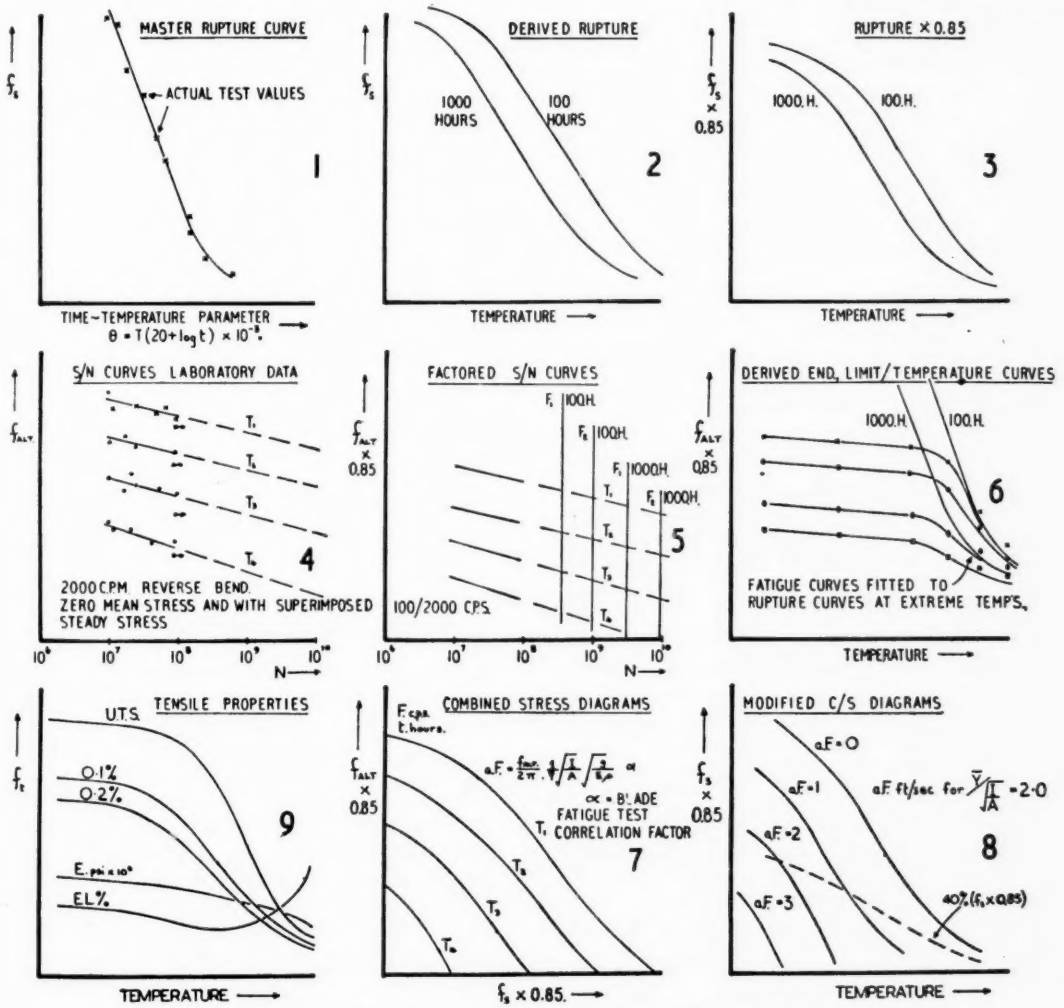


FIGURE 10. Derivation of design data from laboratory mechanical test, and blade fatigue test results.

propose a heat-treatment, in all good faith, which is incorrect for the mechanical duty the blade is to perform. This is illustrated in Fig. 11 where the solid lines show the observed Goodman diagram for Nimonic 90 at 800°C with what we will call a standard heat-treatment. This should be compared with the chain-dotted line representing the modified heat-treatment proposed when the particular duty required was specified as the ability to withstand the steady stresses of approximately 6 tons p.s.i. with the maximum reserve for vibration. While the standard heat-treatment will be seen to equal the modified heat-treatment in pure creep strength (the points on the abscissa) and to be superior to the modified heat-treatment in pure fatigue (the points on the ordinate axis), it is greatly inferior to the modified heat-treatment at the 6 tons steady stress condition. It is thus important for the duty of the blade to be specified exactly when seeking a heat-treatment.

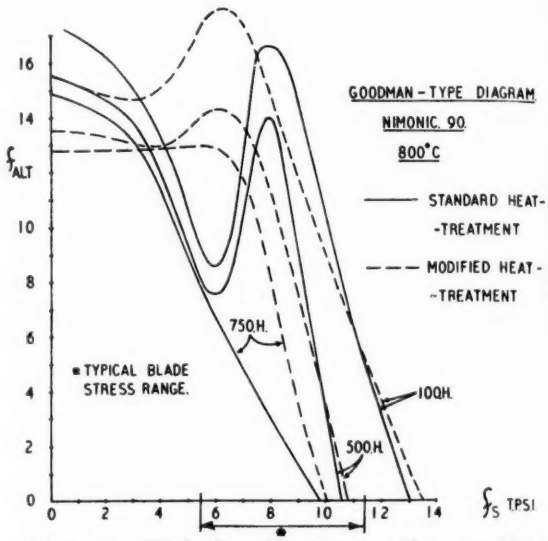


FIGURE 11. Elimination of premature turbine rotor blade failures by heat-treatment modification. Qualitative correlation of laboratory data with engine test results.

5. Quality Control Aspects

If we imagine that we are following the engineering of the blade through from Project Design, it is possible to imagine that the previous considerations have brought us to the point where we are ready to pass an instruction to manufacture to Production. The focal point of this stage is the preparation of a detail drawing and this throws up a new series of problems which are, in the broadest sense, of quality control. The essential points to the next part of the argument are illustrated in the incomplete sample drawing shown in Fig. 12.

5.1. BLADE FORM SPECIFICATION

Specification of blade form has been the subject of much discussion and attempted standardisation in the past, but still remains very much a matter of opinion in practice. Investigation has revealed that there are basically two approaches which can be made to this subject. The first is to say that an unambiguous geometrical definition of shape in purely geometrical terms is all that is required of the blade draughtsman. This approach sticks firmly to the tenet that if suppliers then get into difficulties, it is their own fault and of no concern to the draughtsman. This approach leads to ordinates as a method of specification of blade shape and dimensions to points in "mid-air." The other approach starts with the belief that a detail drawing should continually endeavour to be a direct manufacturing drawing, in the interests of overall economy in time and of minimising human error in making the inevitable interpretation into practical terms.

Adopting the second approach it was clear that the basis of manufacture of blades was a layout of the blade form. It was also established that with the equipment generally existing, and likely to exist for many years to come, layouts drawn from ordinates would differ from each other to an extent which was a significant part of the tolerance allowed. It was found that different sections of manufacture were already having to compare to agreed layouts and were moving towards the idea of a master layout to avoid this in the future. It was, therefore, concluded that the proper thing to do was to draw a *master* layout at the detail stage and call that a part of the blade detail drawing. To do this, a stable drawing material had to be found and printing processes had to be adapted which permitted:

(*a*) The production of exact replicas on stable material in any quantity.

(*b*) The ability to enlarge or reduce from the master by precise means which would conform to the various magnifications of inspection equipment available.

(*c*) Sections to be stacked one on top of the other for one inspection process and to be presented separately for other purposes such as gauge making.

(*d*) Planning Departments to add both notes and further gauge lines, tolerances, shrinkage allowances, and so on, to the original master without undue difficulty.

This problem has been solved and Fig. 13 gives some idea of a typical compressor blade master layout from which all the various requirements of manufacture can be met by a combination of printing and processes, such as enlargement and reduction of scale.

5.2. BLADE DIMENSIONING AND TOLERANCES

Blade inspection is a major factor in determining the ultimate cost of the blade in production and the method of tolerancing is very important in this respect because it can cause unnecessary scrap, due basically to confusion of interpretation. In this connection it is dangerous to assume that blade form tolerances are the only offenders as it has been found that fir-tree root tolerances and platform and shroud tolerances can all be a source of difficulty. Ambiguity is most readily avoided by ensuring that the method of dimensioning

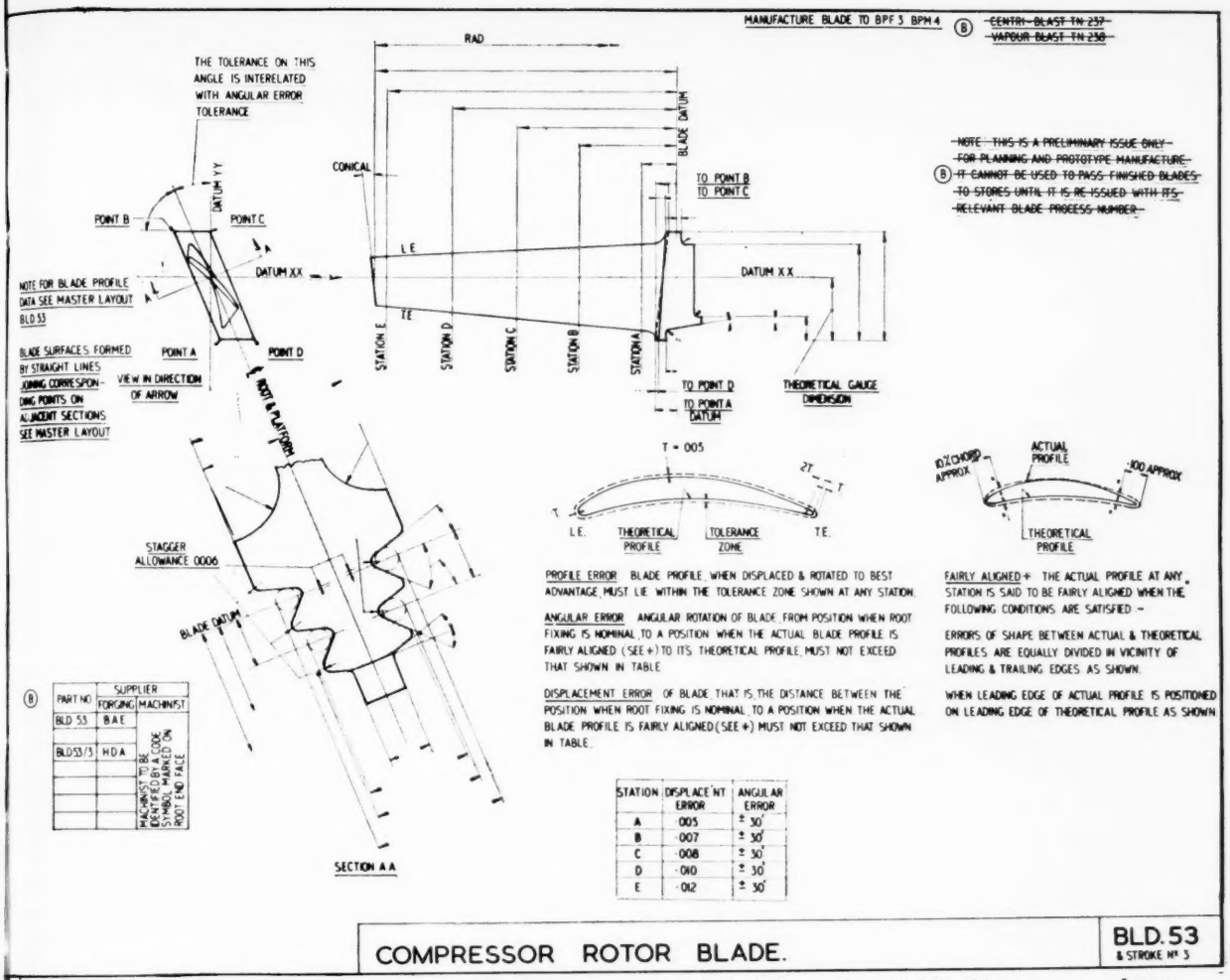

MANUFACTURE BLADE TO BPF 3 BPM 4
(B) ~~CENTRI-BLAST TN 237~~
~~VAPOUR BLAST TN 230~~
~~NOTE: THIS IS A PRELIMINARY ISSUE ONLY FOR PLANNING AND PROTOTYPE MANUFACTURE~~
(B) ~~IT CANNOT BE USED TO PASS FINISHED BLADES TO STORES UNTIL IT IS RE-ISSUED WITH ITS RELEVANT BLADE PROCESS NUMBER~~
THE TOLERANCE ON THIS ANGLE IS INTERELATED WITH ANGULAR ERROR TOLERANCE
RAD
BLADE DATUM
CONICAL
TO POINT B
TO POINT C
DATUM Y.Y.
POINT B
POINT C
DATUM X.X.
L.E.
T.E.
NOTE FOR BLADE PROFILE DATA SEE MASTER LAYOUT BLD 53
BLADE SURFACES FORMED BY STRAIGHT LINES JOINING CORRESPONDING POINTS ON ADJACENT SECTIONS SEE MASTER LAYOUT
POINT A
POINT D
VIEW IN DIRECTION OF ARROW
STATION E
STATION D
STATION C
STATION B
STATION A
TO POINT D
TO POINT A
DATUM
THEORETICAL GAUGE DIMENSION
ROOT & PLATFORM
STAGGER ALLOWANCE 0006
BLADE DATUM
SECTION A.A
T = 005
2T
T
L.E.
THEORETICAL PROFILE
TOLERANCE ZONE
T.E.
10% CHORD APPROX
ACTUAL PROFILE
100 APPROX
THEORETICAL PROFILE
(B)
PART NO
SUPPLIER
FORGING
MACHINIST
BLD 53
BAE
BLD53/3
HDA
MACHINIST TO BE IDENTIFIED BY A CODE SYMBOL MARKED ON ROOT END FACE
PROFILE ERROR BLADE PROFILE WHEN DISPLACED & ROTATED TO BEST ADVANTAGE MUST LIE WITHIN THE TOLERANCE ZONE SHOWN AT ANY STATION.
ANGULAR ERROR ANGULAR ROTATION OF BLADE FROM POSITION WHEN ROOT FIXING IS NOMINAL TO A POSITION WHEN THE ACTUAL BLADE PROFILE IS FAIRLY ALIGNED (SEE +) TO ITS THEORETICAL PROFILE MUST NOT EXCEED THAT SHOWN IN TABLE
DISPLACEMENT ERROR OF BLADE THAT IS THE DISTANCE BETWEEN THE POSITION WHEN ROOT FIXING IS NOMINAL TO A POSITION WHEN THE ACTUAL BLADE PROFILE IS FAIRLY ALIGNED (SEE +) MUST NOT EXCEED THAT SHOWN IN TABLE.
STATION
DISPLACEMENT ERROR
ANGULAR ERROR
A .005 ± 30'
B .007 ± 30'
C .008 ± 30'
D .010 ± 30'
E .012 ± 30'
FAIRLY ALIGNED + THE ACTUAL PROFILE AT ANY STATION IS SAID TO BE FAIRLY ALIGNED WHEN THE FOLLOWING CONDITIONS ARE SATISFIED –
ERRORS OF SHAPE BETWEEN ACTUAL & THEORETICAL PROFILES ARE EQUALLY DIVIDED IN VICINITY OF LEADING & TRAILING EDGES AS SHOWN.
WHEN LEADING EDGE OF ACTUAL PROFILE IS POSITIONED ON LEADING EDGE OF THEORETICAL PROFILE AS SHOWN
COMPRESSOR ROTOR BLADE.
BLD.53
& STROKE Nº 3

FIGURE 12. Compressor rotor blade drawing illustrating blade process control (including re-issue "B" with summary reference numbers), suppliers identification panel, master layout reference for blade shape, blade profile tolerancing and method of dimensioning of blade.

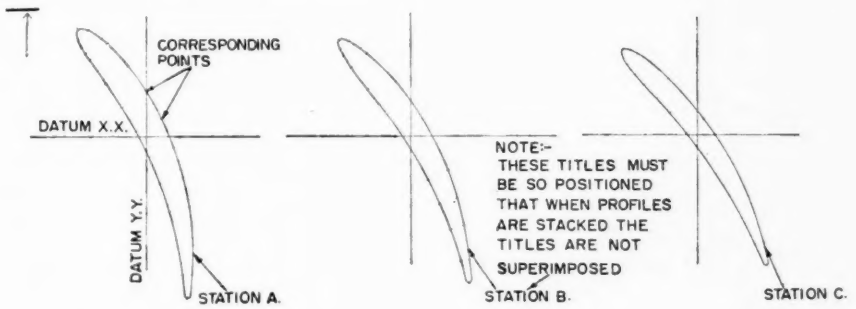
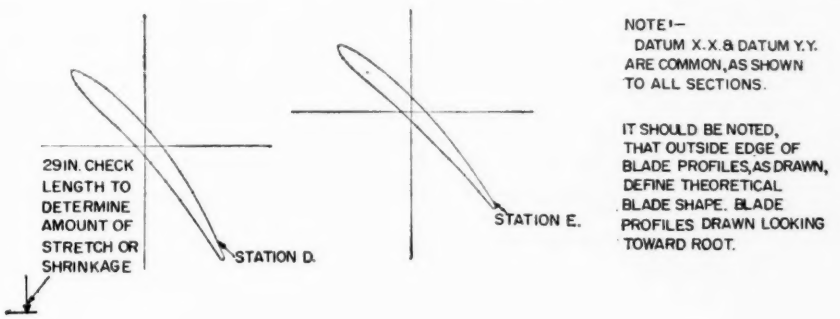


FIGURE 13. Blade layout.


29IN. CHECK LENGTH TO DETERMINE AMOUNT OF STRETCH OR SHRINKAGE
STATION D.
STATION E.
NOTE:-
DATUM X.X. & DATUM Y.Y. ARE COMMON, AS SHOWN TO ALL SECTIONS.
IT SHOULD BE NOTED, THAT OUTSIDE EDGE OF BLADE PROFILES, AS DRAWN, DEFINE THEORETICAL BLADE SHAPE. BLADE PROFILES DRAWN LOOKING TOWARD ROOT.

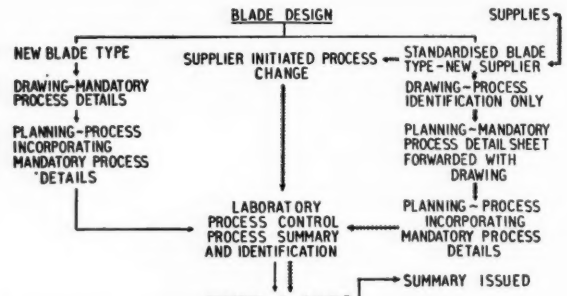


FIGURE 14. Process control.

and tolerancing admits of "first principle" inspection. While there are many ways of inspecting a blade which may be adopted in production quantity inspection, it is agreed practice to prove the method by ensuring that it gives the same answer as a "first principle" (surface table, sine bar, slip gauge, optical projector) method of inspection. If the drawing does not admit of direct first principle inspection then it has to be interpreted and disputes arise.

Figure 12 carries the notes explaining the method of blade form tolerancing which were derived from full discussion with manufacturing representatives with avoidance of ambiguity of interpretation in mind. It also illustrates how other dimensions are specified with this end in view.

The size of the tolerances to be used is another problem. It is not yet clear what are proper blade tolerances, bearing all things in mind, but there seems little to commend the "incentive" tolerancing system, using a difficult target figure to encourage the greatest accuracy possible. There must be at least a 95 per cent probability of all dimensions being within the drawing tolerance for it to be of any use for quality control.

A vital corollary is that the method of inspection should have an inherent error not greater than 20 per cent of the tolerance. This is often the reason for wide tolerances and not the accuracy of the machinist.

5.3. BLADE MATERIAL QUALITY CONTROL

Complementary to dimensional quality control is material quality control. As shown, the designer has already chosen the material he requires and the heat-treatment. The first of these is covered by the company specification which lays down, even for proprietary materials, the agreed chemical composition, method of working and mechanical test procedure. There remain, however, two areas which are not controlled by this method which have emerged as of importance in service. The first of these is the process of manufacture adopted by the Forge Shop, Foundry, and Machine Shop, concerned in the manufacture of the blade. Apparently harmless changes in manufacturing process have resulted in premature failures. To cater for this a blade process control procedure was developed. The procedure takes a different form for different aspects of the subject and is shown diagrammatically in Fig. 14 with corresponding references on Fig. 12. The following main categories of problems arise and are treated in the manner shown:—

(*a*) A new blade type is issued to a drawing which carries instructions on the mandatory aspects of the process, which engineering practice dictates, such as heat-treatment. The drawing also carries a veto forbidding the acceptance of blades into the Stores against this drawing until it has been raised in issue and a process summary reference added. This enables the manufacture of the blade to proceed to any process which the manufacturer desires, but makes it mandatory that a summary of the process, taken from the operation sheets, shall be prepared by the Laboratory Process Control section and a reference added to the drawing before the blades are accepted as finished. In rare instances it may be that the laboratory reject the blades due to an undesirable feature of the process, but this is unlikely as the closest liaison is maintained between Planning Departments and the Laboratory Process Control. This new blade then proceeds to development engine test and may eventually find itself in a type test engine or modification approval engine and be given a production clearance.

(*b*) We now come to the second category of blade process control which can, for example, arise after the blade has been standardised for production. The supplier of the blade may say that this development blade has been made by a method which he would not adopt for large quantity production and he then requests permission to change his process, usually with a justification based on price or quantities required for the engine programme. This then proceeds down the centre track shown on Fig. 14, except that laboratory assessment may now do one of three things. It may reject the permission for the change proposed and re-open negotiations with the supplier. It may accept the process on entirely theoretical assessment as representing harmless minor changes, incorporate the changes as agreed alternatives in the original process summary, and re-issue that summary. Finally, it may find insufficient evidence to justify straight acceptance or rejection and demand that the change be put to engine development test. A new drawing is then issued carrying a separate part number. Blades are made to this drawing with a new process number and submitted to the necessary rig and engine tests. Subject to satisfactory performance, these blades are then standardised for production.

As a result of this procedure, it is possible to say that new suppliers can be brought in to assist production or to permit the manufacture of engines under licence overseas in a perfectly rational manner with sound quality control safeguards.

The final factor in quality control is the supervision

under which the blade is made to all the instructions implied on the final detail drawing. To this end the supplier is identified with a stroke number after the part number, implying full interchangeability, permission to store blades in one bin with a variety of stroke numbers, and so on, but allowing the identification of the supplier to be available for failure investigations.

The aim and object of this has been to remove the usual supply identification as the dominant factor in quality control, since having identified the supplier the hazard is still left that he may change his process to the detriment of the finished article. This is made subsidiary to a system aimed at precluding significant changes being made without their being evaluated by the Laboratory and recognised on the detail drawing. This enables Production Control to have increased flexibility of sources of supply while still safeguarding the engineering quality of the blade.

Figure 15 shows the results of rig tests which have been made to answer material quality control questions. It will be noticed that actual blades have been used and the fatigue results have all been presented in $a \times f$ form. They are all self explanatory and are presented as an indication of the problems arising in this field.

6. Concluding Remarks

This paper has endeavoured to show a sample of the work which has been done to examine the mechanism of failure of blades, and the allied subjects of blade materials and quality control. It is hoped that it will stimulate further papers in which other work in this field will be made available. Only in this way can we all hope to build up a body of knowledge adequate to tackle the problems the future undoubtedly has in store.

ACKNOWLEDGMENTS

The author wishes to express his gratitude to the

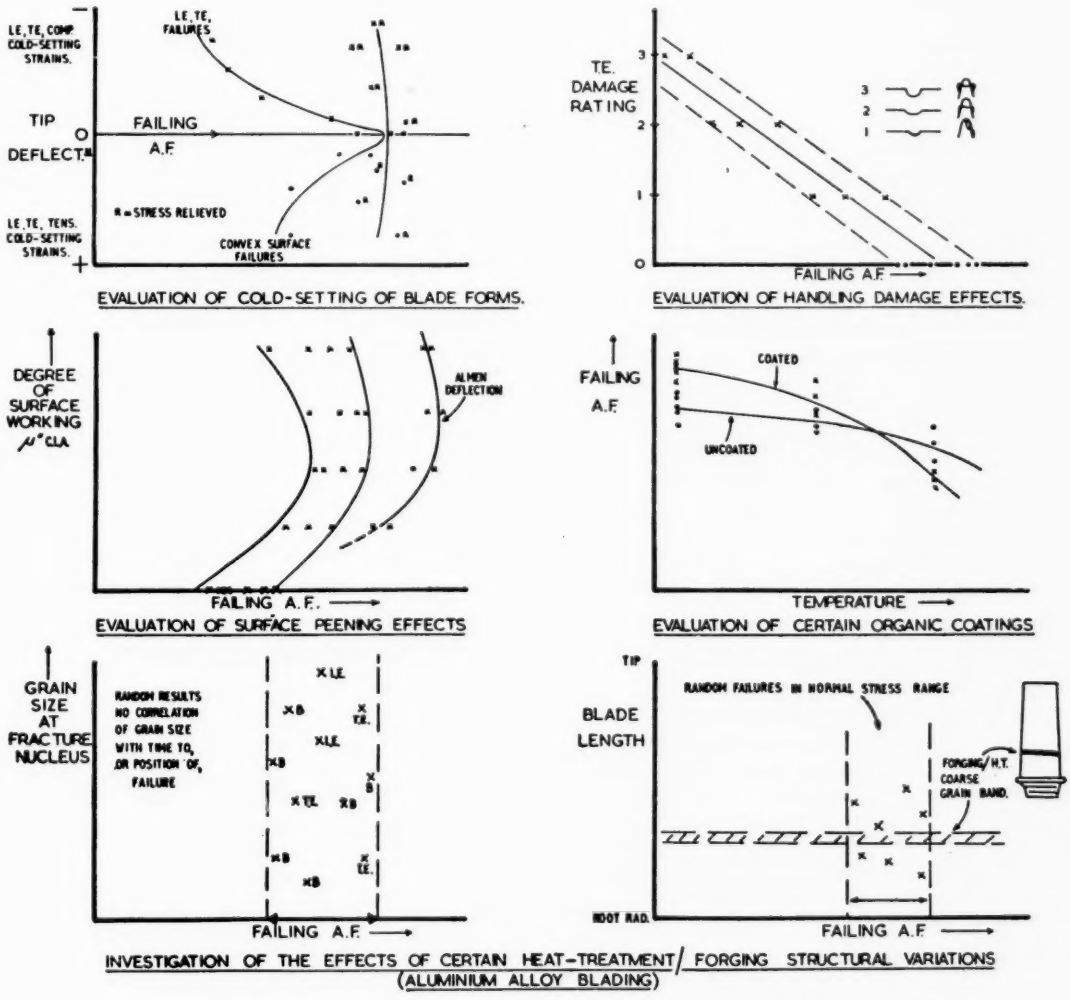


FIGURE 15. Typical blade fatigue tests undertaken to investigate some aspects of blade processing.

Directors of Bristol Aero-Engines Limited for permission to publish this work. He would also like to acknowledge the co-operation of many of his colleagues in the Company and in particular the work of collaborators Drs. J. Swirles and E. K. Armstrong and Messrs. J. R. D. Fuller, C. Morris, W. S. Pollinger, T. F. Blenkiron and L. A. McGee (attached from Australian Aeronautical Research Laboratories, Fishermen's Bend) who constituted the research team.

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Gas Turbines For Helicopters

by

A. W. MORLEY, Ph.D., A.F.R.Ae.S.
(*Project Engineer, D. Napier and Son Limited.*)

1. Introduction

Rapid developments in the use of the helicopter were made in the Korean War. These were taking place at the time when there was considerable urge to introduce the gas turbine engine into the civil aircraft market. It was also a time when much effort was being expended on various forms of reaction propulsion, mostly for missiles. A number of new helicopter projects were started, taking advantage of the new knowledge in propulsion engineering; some utilised new gas turbine designs and others various forms of tip jet reaction. Liquid fuel rockets, ram-jets, pulse-jets and air pressure jets were tried. However, the main line of development continued to be the gradual improvement of direct mechanical drive.

Since Korea the natural trend has been first to convert existing helicopters to turbine engines. When new helicopters were designed wisdom dictated mechanical drive pending the gathering of sufficient experience with other forms of rotor power, and so far the accumulated knowledge of the mechanical drive has proved of greater value to helicopter progress than the potential advantages offered by the alternative engine forms.

Most existing helicopters, leaving out new turbine-driven types, use a constant rotor speed over a wide range of operating conditions. This technique has come from the marrying up of the characteristics of the piston engine with the control characteristics of the helicopter rotor; it does not necessarily give best results with a gas turbine.

The turbine for mechanical drive has been called a "turboshaft" engine. It may be fundamentally either "single-shaft" or "free turbine." The single-shaft engine needs a clutch and freewheel to separate the engine and rotor during starting and auto-rotation. The free turbine dispenses with the clutch entirely. The elimination of the clutch, and also the cooling fan required by the piston engine, results in considerable simplification and saving in space and weight of the power unit. Thus the installed weight of a free turbine engine is roughly one-half to two-thirds that of the corresponding piston engine. This saving leads to all-round improvements in performance and to the possibility of providing an economic twin-engined helicopter which has sufficient power to ensure level flight on one engine.

Much data has already been published on the various helicopter gas turbines and their advantages and problems. Table I gives a list of the notable applications so far and the references attached at the end of the paper will enable a selection of the available information to be found. Napier experience up to the present time has been principally with a fixed shaft engine, the Eland, used as a pressure jet engine for the Fairey Rotodyne and as a mechanical drive engine in the Westland Westminster; also with a free turbine engine, the Gazelle, used in the Westland Wessex and Bristol 192. Some work has been done to design and complete the type test of a mixed jet engine, the Oryx, but so far no helicopter using this has got beyond the stage of initial rotor trials.

2. Application of Gas Turbines to Mechanical Drive

In a fixed wing turbo-propeller type the engine is the main item and dictates the propeller design so that it is regarded as part and parcel of the power plant. In the helicopter the rotor is not only the wing system, it is also the main aerodynamic control, as well as the component which absorbs the engine power. With the rotor head it represents a far more expensive part of the whole than a propeller. As a rule it cannot be tailored to suit the engine; the engine, more than likely, must be tailored to suit it. For this reason the free turbine is well placed, for it has more flexibility to match in with rotor requirements than the fixed shaft engine. With a lucky choice of reduction gear a fixed shaft

TABLE I

Helicopter and Country of Origin	A.U.W. lb.	Rotor Data: Drive	Rotor Data: Configuration	Rotor Data: Disc Loading lb./ft.^{2}	Power Unit and Power	Power Loading	Remarks
Nord 1750 Norelfe France	1,892	Shaft Drive	1 × 3B	1·845	1 × Turboméca Artouste 320 s.h.p.	5·88 lb./s.h.p.	2-3 seats. Jet torque control. Origin of Aerotechnica AC-13 and -14. Apparently discontinued 1957.
Aerotechnica AC-13A Spain	1,962	Shaft Drive	1 × 2B	2·86	1 × Turboméca Artouste I 260 s.h.p.	7·54 lb./s.h.p.	3 seats. Swivelling tail pipe for directional control and thrust.
Aerotechnica AC-14 Spain	2,650	Shaft Drive	1 × 4B	3·40	1 × Turboméca Artouste II 460 s.h.p.	6·625 lb./s.h.p.	5 seats. Development of AC-13A. Similar jet torque control and thrust augmentation.
Bell 201 U.S.A.	—	Shaft Drive	1 × 2B	—	1 × Continental T51-T-3 (Artouste) 360 s.h.p.	—	2 seats. Developed from Model 47. Army utility.
S.E. Alouette II France	3,300	Shaft Drive	1 × 3B	3·74	1 × Turboméca Artouste II 400 s.h.p.	8·25 lb./s.h.p.	2 seats. Developed from piston-engined Alouette I. Military and civil applications.
Sikorsky S.59 U.S.A.	3,650	Shaft Drive	1 × 4B	3·88	1 × Continental T51-T-3 (Artouste II) 400 s.h.p.	9·12 lb./s.h.p.	4 seats.
Bell 204 U.S.A.	5,620	Shaft Drive	1 × 2B	3·70	1 × Lycoming T.53 825 s.h.p.	6·82 lb./s.h.p.	4 seats. Army utility.
Kaman HU2 K-1 U.S.A.	—	Shaft Drive	1 × 4B	—	1 × G.E. T.58 950 s.h.p.	—	Naval applications.
Westland Wessex G.B.	12,600	Shaft Drive	1 × 4B	5·12	1 × Napier Gazelle N.Ga.13 1,450 s.h.p.	8·69 lb./s.h.p.	14 seats. Adapted from S.58.
Sikorsky S.61 U.S.A.	—	Shaft Drive	1 × 4B	—	2 × G.E. T.58 1,900 s.h.p. total	—	16 seats. Re-engined S.58. Commercial uses.
Westland Westminster G.B.	33,000	Shaft Drive	1 × 5B	8·11	2 × Napier Eland N.El.1 4,800 s.h.p. total	6·88 lb./s.h.p.	39 to 43 seats. Also flying crane version.
Mi 6 Russia	60,000 (app.)	Shaft Drive	1 × 5B	—	2 × Soloviev 7,000 s.h.p. app.	8·27 lb./s.h.p. (app.)	60 to 70 seats.
S.O. Djinn France	1,765	Tip Drive Cold Pressure jets	1 × 2B	1·735	1 × Turboméca Palouste 250 g.h.p.	7·06 lb./g.h.p.	2 seats. Military and civil applications.
Fairey Ultra-Light G.B.	1,900	Tip Drive Hot Pressure jets	1 × 2B	3·02	1 × Blackburn Palouste 252 g.h.p.	7·54 lb./g.h.p.	2 seats. Army A.O.P.
Fiat Model 7002 Italy	3,087	Tip Drive Cold Pressure jets	1 × 2B	2·53	1 × Fiat 4700 530 g.h.p.	5·83 lb./g.h.p.	7 seats or equivalent payload. Free turbine engine driving air compressor.
Hunting P.74 G.B.	7,750	Tip Drive Pressure jets	1 × 3B	3·33	2 × Napier Oryx N.Or.1 1,500 g.h.p. total	5·17 lb./g.h.p.	Research helicopter. Pressure jets from mixed turbine exhaust and auxiliary compressor.
Fairey Rotodyne G.B.	39,000	Tip Drive Hot Pressure jets	1 × 4B	6·14	2 × Napier Eland N.El.7 6,500 s.h.p.* total	8·5 lb./g.h.p.	42 seats. *Power delivered to auxiliary compressors at take off.
Hughes U.S.A.	46,500	Tip Drive Hot Pressure jets	1 × 2B	3·52	2 × Allison J.35 10,000 lb. total	—	2 seats. Flying crane for U.S.A.F.
Vertol H.21 (Experimental) U.S.A.	16,200	Shaft Drive	2 × 3B Tandem	5·30	2 × Lycoming T.53 1,650 s.h.p. total	9·82 lb./s.h.p.	Prototype. 22 or 10 seats.
					2 × G.E. T.58 1,900 s.h.p. total	8·52 lb./s.h.p.	Army prototype.
Bristol 192C G.B.	18,000	Shaft Drive	2 × 4B Tandem	4·84	2 × Napier Gazelle N.Ga.2 2,600 s.h.p. total	6·92 lb./s.h.p.	25 seats.
Bell D.216 U.S.A.	—	Shaft Drive	2 × 2B Tandem	—	3 × Lycoming T.53 2,475 s.h.p. total	—	27 seats. Project stage.
Vertol PV-15 U.S.A.	32,000	Shaft Drive	2 × 3B Tandem	3·04	2 × Allison T38-A-6 6,000 s.h.p. total	5·34 lb./s.h.p.	42 seats. Originally Piasecki YH-16A.
Doman-Fleet Commercial Project Canada	21,000	—	—	—	3 × G.E. T.58 2,850 s.h.p. total	7·37 lb./s.h.p.	20 seats.
Hiller HJ-1 U.S.A.	1,080	Tip Drive Ram-jets	1 × 2B	2·61	2 × Hiller 8RJ2B 90 rotor h.p. total	12·0 lb./r.h.p.	2 seats. Civil model Hornet.
Kolibrie H-3 Netherlands	1,440	Tip Drive Ram-jets	1 × 2B	1·685	2 × TJ-5 ram-jets 110 rotor h.p. total	13·1 lb./r.h.p.	2 seats.

engine may happen to suit the rotor and with constant r.p.m. technique may turn out to be fully satisfactory. But when full advantage is taken of the greater flexibility of the free turbine, particularly in multi-engine machines which must fly safely after engine failure, improvements in helicopter performance and handling are obtainable.

3. Differences Between Rotary and Fixed Wing Turbo-Shaft Engines

Features which ascribe to the suitability of the gas turbine for a mechanical rotor drive may be noted by reference to the major differences in philosophy between rotary wing and fixed wing turbo-shaft power plants. (Table II.)

Designers of turbo-shaft engines looking for applications to both helicopters and turbo-props, are obliged to make a careful study of all possible drive arrangements. For the helicopter the lightest engine is believed to be one in which the drive is taken from the rear of the power turbine, through a reduction gear supported on the exhaust casing, to a shaft capable of being extended in any direction to suit the aircraft. In small sizes of free turbine engine, the alternative of taking the drive shaft through the centre of the main rotor to a nose reduction gear means a slightly larger rotor shaft diameter. Although a front drive is ideal for turbo-propellers it is likely to be slightly heavy for the helicopter.

The helicopter rotor drive is always near vertical; so the most compact arrangement will be with the engine drive shaft vertical, provided that the gas turbine can run this way without undue cost. This dictates the configuration in which the engine intake is at the bottom of the unit to avoid the risk of fire from a false start. The power turbine will then be uppermost with the drive coming from the rear. The reduction gear must be placed adjacent to the power turbine, conveniently between the branches of the exhaust system. Further, the turbine must be designed to give the minimum residual exhaust thrust and the exhaust gases must be discharged horizontally to prevent any downward thrust component. Such an installation, suitable for continuous vertical operation, will add a little weight to the engine principally on account of drainage problems which do not occur in the horizontal engine. With due care this extra weight can be much less than would be added by a bevel gear to convert the drive from the horizontal to the vertical.

In practice the helicopter engine may be required to suit any position from horizontal to vertical. This will demand considerable detail modification to an engine previously used for fixed wing applications, where the axis is restricted to the near horizontal. Obviously the stressing cases are different, for vertical g becomes an axial acceleration instead of transverse. Such engines are not usually cleared for more than a few seconds of vertical running whereas for the helicopter engine, to suit any position, the lubrication and bearing arrangements must be so designed that a type test can be completed at any inclination.

TABLE II

	Turbo-Prop Requirement	*Helicopter Mechanical Drive Requirement*
Type test clearance:	1. Take-off	1. One hour
Rating duty	2. Continuous	2. Continuous 3. Emergency overload
Power take-off	Front	Rear. All directions.
Direction of axis	Horizontal	Vertical to horizontal
Intake arrangements	Forward facing for ram	Static chamber
Gear drive	High ratio, short drive, direct reduction to propeller r.p.m.	Low ratio, long or short transmission. Second gear box before rotor head
Controls	Throttle used for thrust control C.S.U., fuel governor, and torquemeter linked	Throttle used for flying control in conjunction with rotor pitch. Torquemeter controls emergency power of other engine in twin installation

3.1. INTAKE CONDITIONS

Secondly, the intake conditions are quite different between the two classes of power unit. The forward speed of the helicopter is too low for intake ram to influence the design, and the axial intake of a jet engine is poor for a vertical installation. A plenum chamber with side intakes is probably the best arrangement, as it can be made to give uniform entry flow over the compressor face, whatever the angle of the engine axis relative to that of the aircraft.

The intake must be protected against long periods in low altitude weather, including low flights over water. The potential icing risk is likely to be great. On the other hand the engine need not be exposed, but may be tucked well inside the fuselage where the parts susceptible to icing can be supplied with waste heat.

The intake must also be proof against debris swept up from the ground by the rotor, or by any other aircraft in the vicinity. Near industrial towns atmospheric pollution is heavy and, with the high air consumption of the gas turbine, a considerable weight of dirt reaches the intake. Compressor performance must not be seriously affected. Again the exhaust must not be allowed to recirculate back into the intake for it might cause a tendency to compressor surge on opening up, or bad loss of power due to high intake temperature. When all these requirements are satisfied the intake must still have a low pressure loss and a good circumferential pressure distribution, both when running on the ground and in flight.

3.2. REDUCTION GEAR

Thirdly, the output r.p.m. of the helicopter gas turbine engine will be higher than that of the corresponding turbo-prop. Mechanical drive of helicopter rotors has evolved from existing piston aero-engines, with the aircraft designer providing gear boxes for the rotor as necessary. The helicopter gas turbine engine usually only caters for the high speed section of the overall speed reduction. A typical gear ratio of the

Gazelle, for example, is 6 to 1, which is followed by a rotor gear box reduction ratio of the order 10 to 1, making 60 to 1 overall reduction from power turbine shaft to helicopter rotor. Current American practice uses an even lower ratio for the engine gear, for example, 3 to 1. This helps the engine designer to achieve a low engine weight, but only at the expense of the rest of the rotor gear box weight. Where there is an extended transmission from the engine to the rotor it may appear better to put the major speed reduction at the rotor end. Whirling may then become a problem making a stiffer shaft necessary. This will offset the gain due to the reduced twisting moment.

When a fixed shaft engine is to be used for a mechanical drive there is little choice of the reduction gear ratio, since this is dictated by the rather narrow range of engine r.p.m. over which rotor performance will be optimum. The steepness of the power curve of the fixed shaft engine is such that it can only be matched for peak output at peak r.p.m. On the other hand the free turbine engine offers considerable scope for selection of best r.p.m. This is principally because of its flat output power curve which also gives it the desirable negative slope torque characteristic. Over a wide range of output speed the power changes only slightly making the choice of reduction gear for a given helicopter rotor immaterial as far as the power is concerned. It is quite possible to design the power turbine for a lower r.p.m. than the compressor turbine, in order to save weight overall on the free turbine and reduction gear.

The engine reduction gear ratio has an effect on the rotational inertia of the engine rotor system. For example, if the power turbine r.p.m. is made lower so that the engine reduction gear weight for a given output r.p.m. is reduced, the rotor system will add more inertia to the power turbine. This will tend to slow up the response of the engine output shaft to movements of the pilot's lever for changes of load if accompanied by changes in output shaft or rotor r.p.m. The free turbine engine can be better in respect of total moment of inertia even when the effect of the rotor inertia is included, than the fixed shaft engine of corresponding size, although the latter, as is most likely, has the higher reduction gear ratio.

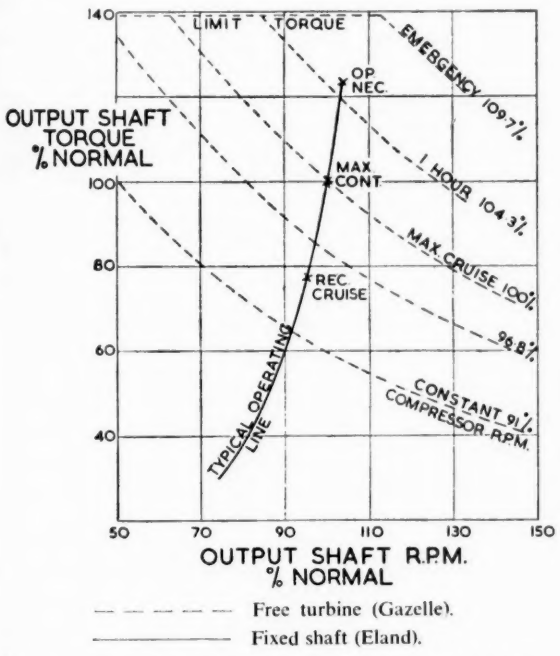

FIGURE 1. Torque characteristics and ratings of free turbine helicopter and fixed shaft turbo-propeller engines.

3.3. CONTROL OF ENGINE

Present helicopter engine control systems, with a free turbine engine, depend upon the pilot maintaining rotor r.p.m. while the engine is governed automatically to a selected compressor r.p.m. The pilot has a single lever engine control which may be interconnected with his collective pitch control in much the same way as propeller pitch is interconnected with the engine control of a turbo-prop engine. This interconnection may be such that at a specified condition of flight, *e.g.* vertical climb, the power output of the engine is adjusted automatically with rotor collective pitch to maintain constant rotor r.p.m. A trim can be provided in the form of a twist grip on the pilot's pitch control stick to permit adjustments for various flight conditions. In this way the engine will handle according to accepted techniques.

With the free turbine engine the output torque increases if the resistance to rotor rotation increases, causing its r.p.m. to fall off and vice versa. (Fig. 1.) This negative slope torque characteristic helps to restore any unselected fall-off in rotor speed brought about by added resistance to rotation and gives stable rotor speed control. With constant rotor r.p.m., changes in power can be made very rapidly, if necessary, because only the gas generator has to change speed. On the other hand the fixed shaft engine has a positive torque characteristic and there is only a narrow r.p.m. range over which the power can be changed without sacrificing fuel economy.

To take full advantage of the great flexibility of the free turbine engine the compressor must have an ample surge margin; also the gas generator must be fully protected against over-speed and under-speed. The rates of acceleration and deceleration must be limited automatically to a safe range. In handling the engine the pilot then needs only to watch rotor r.p.m. and trim accordingly.

At the expense of added complication to the engine controls a second governor can be added to take automatic charge of rotor r.p.m. (except for auto-rotation). Such a control has been described in accounts of American helicopter gas turbine units[5], but has not so far been adopted for British engines.

With twin-engined helicopters which are required to fly level on a single engine, extra complications are worth adding to the automatic control. If engine failure occurs it can be arranged that the remaining engine gives more than its normal maximum power for a short period to enable the pilot to maintain height.

Depending upon operational requirements the permissible delay between the failure of the one engine and obtaining the overload power of the other, must be very short, for example two to three seconds. A torque-meter can be incorporated in the engine gear box and signals from the torque-meter fed to the control system so that the failure of one engine automatically brings about a rapid increase in power from the other. The same torque-meter can be used to give protection against engine damage from excessive output torque at low rotor r.p.m. During the period of engine overload the control must keep a strict limit on the turbine inlet temperature, and this can be done by feeding in to the control a signal from a turbine inlet temperature measuring device, which is biased upward for the emergency.

Another important point in a twin engine installation is that in no way must it be possible for the safe maximum rotor power to be exceeded, by both engines accidentally giving their emergency power at the same time. The control must ensure that the output of one engine does not exceed half the safe rotor maximum by more power than the second engine is short of it. A practical way of achieving this with safety, is to operate the boosting circuit of the live engine by a switch energised when the fuel dump valve opens on the failed engine.

4. Installation of Gas Turbine in Helicopter (Mechanical Drive)

Maximum benefits in installation efficiency are obtained when the engine is designed specifically for a given helicopter, but as this is not economically feasible a compromise has to be made by designing the unit to be adaptable. With the actual layout in the aircraft uncertain, the tendancy is for the engine to be made as compact and light as possible by separating off as much of the transmission and other weight as can properly be associated with the airframe. This leads to some confusion over what is to be included in the engine weight and what is not. It is not unusual to find that as much as 20 per cent is shorn off the gross weight of the power unit in this way.

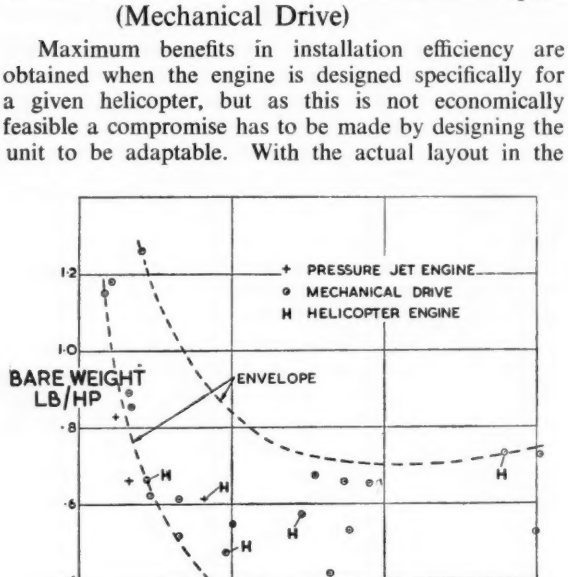

FIGURE 2. Specific weights of small turbo-shaft engines from published data based on s.h.p. or g.h.p. for helicopter units and e.s.h.p. for turbo-props.

Weight figures of helicopter units are particularly difficult to compare. Besides the trouble over transmission weight, one sometimes finds it difficult to assess how much of the control system is quoted as engine dry weight, and whether all essential accessories have been allowed for. In a full comparison of helicopter gas turbine weights the provision for anti-torque must be debited, since to leave this out penalises the gas drive for which no anti-torque gear is required. Again there is the allowance for maturity. A new engine must be on the weight minimum to be attractive, but experience shows that weight climbs as development proceeds. These points must be borne in mind when considering the collected data shown in Fig. 2. This figure gives an idea of the trend of specific weight with size and indicates that the minimum weight per horse power is reached between the 1,000 and 2,000 h.p. size.

The mechanical drive helicopter engine lends itself to installation in the form of a complete power pack or engine change unit. To simplify the drive the unit is usually installed well inside the fuselage where it is difficult to get at for maintenance. It is, therefore, necessary to make careful provision for its easy removal. This is particularly so with a vertical engine buried underneath the rotor head, where the only practical way out is downward through the body of the aircraft. The buried power unit must be enclosed in a cell separated into zones by fireproof bulkheads which isolate the hot turbine and exhaust duct areas from the rest of the aircraft. A fan is required for the oil cooler and this fan can be engine-driven and made large enough to ventilate the zones and supply essential cooling to accessories at the same time. The eduction effect of the exhaust gases can be used to assist the ventilation of the hot bay.

Although cooling is no longer the problem it was with the enclosed piston engine, the fan demand tends to increase as equipment is added to the aircraft in service for there is no ram ventilation. The worst case for fan load is emergency power under tropical conditions.

There is less risk of fire from a false start if the exhausts are situated above the combustion chambers, for then any unburnt fuel will be blown out through the exhaust, which should take it clear of the aircraft. As the compressor takes some seconds to slow down with the fuel turned off, no fuel can drain through to the bottom of the engine. The same applies to any residual fuel on shut down.

4.1. THE GAZELLE HELICOPTER TURBINE ENGINE

The Napier Gazelle is a free turbine turbo-shaft unit intended for helicopter mechanical drive. It is designed as a complete power pack to operate

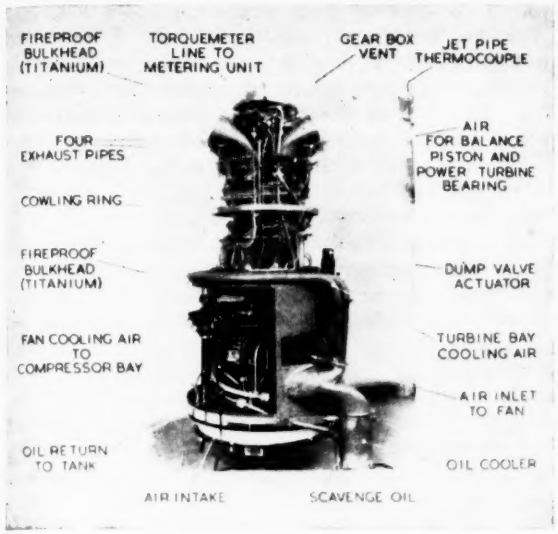

FIGURE 3. The Gazelle.

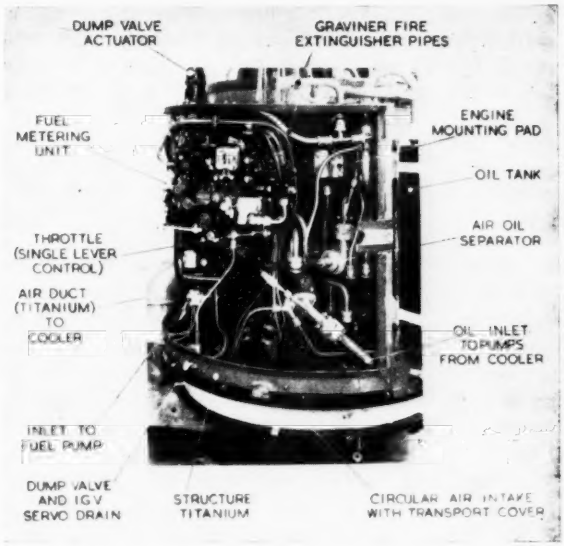

FIGURE 4. The Gazelle.

continuously at any angle from the horizontal to the vertical with the power take-off at the turbine end. For single engine application in the Westland Wessex, where it takes the place of the 1,525 h.p. Wright R-1820 engine of the Sikorsky S-58, it has a maximum one hour rating of 1,430 h.p. It gives the Wessex an enhanced rate of climb and hovering ceiling, despite items which have been added to "anglicise" the S-58, as well as a better payload and cruising speed. The limiting range is reduced however, due to the increased rate of fuel consumption. For the twin-engined Bristol 192 the Gazelle has a 2½ minute emergency rating of 1,650 h.p. to meet the single engine failure case, with a normal maximum one hour power of 1,300 h.p.

The Gazelle is available for either direction of rotation. This is an advantage derived from a free power turbine; only the blade angles of the single power wheel stage and some oil jets in the gearbox need a change from left to right hand to reverse the direction of rotation of the rotor.

The dry weight of the complete engine is 830 lb. (to the official schedule EDM No. 25) while the full power unit, with oil tank, mounting structure, ventilation and fireproofing weighs about another 75 lb. The pressure ratio is about 6·6 at full compressor r.p.m. (20,400). A typical reduction gear ratio is 6 to 1. The Gazelle has already been well described in the Technical Press[8]. Some of the prominent features of the engine in power pack form are indicated in Fig. 3 and, for illustration of the engineering work involved, Fig. 4 gives a closer view of exterior detail in the region of the compressor bay. Considerable use is made of titanium in the formation of the bulkheads and air ducts.

4.2. THE WESTMINSTER HELICOPTER

A new application of mechanical drive is seen in the Westland Westminster, a 33,000 lb. helicopter now being built by Westland from the basic Sikorsky S-56. Two Eland E229 type N.E1.3 fixed shaft engines are to be fitted side by side facing forwards on the roof of the fuselage in front of the main rotor gearbox. The drive from each engine is taken by a separately supported flexible shaft, coupled at the front to the extended turbine shaft and at the rear to a hydraulic clutch. (Fig. 5). The clutch connects to a 3 layshaft spur gearbox of 3·84 step down ratio, which also incorporates the torque-meter. The gear is connected in turn, through a free wheel, to the main rotor gear box. The flexible shaft can be moved a sufficient distance axially to permit the clutch and gearbox to be removed if necessary. The engine installation makes the greatest possible use of parts already developed for other applications of the Eland; thus the propeller reduction gear, which is not now required, provides most of the parts for the 3·84 step down gear.

The rotor gear box, which is an aircraft part, takes the two engine drives, converts them to the vertical by bevels, couples them to a common shaft and, by a final reduction, brings the speed correct for the rotor drive. The overall speed ratio, engine r.p.m. to rotor r.p.m. is about 72 to one.

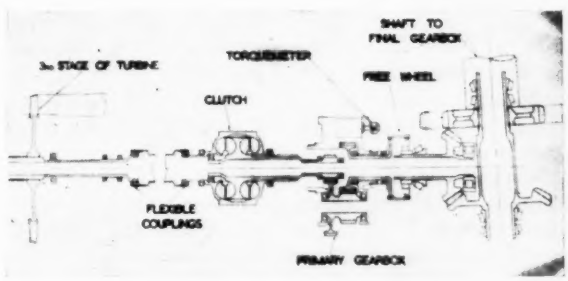

FIGURE 5. The Eland flexible shaft.

The two engines are coupled mechanically when the hydraulic clutches are full and are then controlled as a pair by pilot's selection of the governing r.p.m. In case of any inequality between the two loads the pilot can trim the engines separately, using the two torquemeter readings as a guide. If one engine should fail the governor increases the torque of the other engine to the allowable limit.

5. Application of the Gas Turbine with Gas Drive

Studies such as those described in Reference 9 suggest that a gas drive to the rotor blade tips can sometimes be attractive, in comparison with the direct mechanical drive, particularly for helicopters which are to spend a greater proportion of their flight time in the hovering condition. The gas drive has a somewhat greater cruising fuel consumption per rotor horse power and, as a result, is less attractive overall for long periods of cruising flight. A particularly good case for the gas drive can be made where it is the means of climb and descent but is not used for cruise, as in the Fairey Rotodyne. The simplicity resulting from the absence of a mechanical transmission system, reduction gear and rotor torque reaction permits a neat construction of the rotor head and can yield a helicopter of good load lifting capacity. Interesting small helicopters, such as the Djinn SO.1221 and Fiat 7002, are examples of low cost aircraft in this class.

The successful helicopter gas drives so far, use compressed air from the engine or air from a separately driven compressor. The air flows through passages in the hollow rotor blades to the blade tips. Some rotor power is used in compressing it as it passes through the blades. In the small helicopter, where maximum simplicity is essential, the air is usually discharged from the blade tips into the atmosphere as air propulsive jets; but in other cases the air is first used to burn fuel in small combustion chambers built into the tips of the rotor blades. For maximum rotor power a separate compressor is essential and the highest possible combustion temperature is required (for example about 2,200°K), although the specific fuel consumption deteriorates with high temperatures, and practical difficulties increase. A typical limit is about 2,000°K. The rotor power is varied by the amount of fuel burnt, and there is a roughly linear relationship between combustion temperature and rotor power per lb. of air over the running range.

Increase in ambient air temperatures will bring about a significant loss in performance. This loss will be of the same order as the fall off in power with a turbo-shaft engine. Mass flow and delivery pressure to the rotor both drop and will cause a typical loss in rotor power of 20 per cent for a 30°C increase in atmospheric temperature. Such a loss could be regained by allowing a margin on the tip burner temperature under I.C.A.N. conditions; about 450°C would be required to regain 20 per cent power.

The air supply pressure must be chosen with reference to the mass flow required, the compressor work to be done, the effect of losses in the air ducts and rotor blades, and the practical problems of burning in the high g field. A typical sea level rotor supply pressure is four atmospheres but performance is reasonably good at two atmospheres because the efficiency of the jet improves at low discharge velocities and helps to offset the lower expansion ratio of the thermal cycle.

5.1. ORYX GAS GENERATOR

Another attractive pressure jet system was explored by the development of the Napier Oryx Engine[10, 11]. The Oryx was a form of small by-pass engine designed to power the rotor of the Percival P.74 Helicopter by gas drive. The air from the by-pass compressor was mixed with the exhaust from the main turbine, the whole output from the cycle thus being delivered to the rotor in the form of gas horse power. In this mixed jet system the pressure ratio which could be developed for take-off I.C.A.N. sea level was of the order of 1·6 to 1 and the gas temperature of the order 400°C. These conditions of the gas were acceptable, although not ideal from the point of view of the rotor, but were about the best that was possible using uncooled turbine blades. Much valuable experience was obtained with the Oryx in the P.74, but the experiment did not reach the flight stage due to the cost of developing another rotor when the first proved deficient in lifting capacity.

5.2. ROTODYNE SYSTEM

Further application of the pressure jet system has been undertaken by the Fairey Company with their Rotodyne aircraft[12]. This machine, the general arrangement of which is well known, has two Eland engines arranged so that they can transmit power either

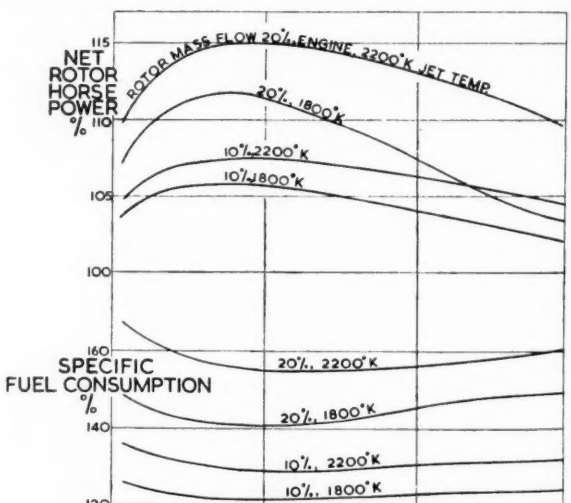

FIGURE 6. Performance of combined mechanical and gas drive. Curves show estimated net rotor horse power and specific fuel consumption as percentages of their values for all-mechanical drive. Rotor tip speed 650 ft./sec.

forward to the propellers, or rearward to auxiliary compressors. The latter supply compressed air to the rotor at a pressure ratio of about 4:1 for the hovering phase of the aircraft. Lift is developed on a fixed wing in forward flight as in an orthodox aircraft. The auxiliary compressor of each engine is coupled to the rear of the turbine through a hydraulic clutch. The clutch is under the pilot's control and can be emptied when the rotor is required to free-wheel for cruising flight. In vertical flight when the clutches are full, the compressor takes practically all the engine power, but sufficient propeller thrust remains for yaw control of the aircraft by the pilot's adjustment of the propeller pitches. For the auxiliary compression ratio of about 4:1 the Rotodyne develops about 115 h.p. in the rotor per lb. of air per second at a specific fuel consumption of 1·6 lb. per rotor h.p. per hr.[14] This consumption refers only to the fuel used in the tip burners and corresponds to a rotor tip speed of 700 ft./sec.

5.3. COMPOSITE SYSTEMS

In the Rotodyne the power plant combines the double function of providing propeller thrust for forward flight and rotor gas horse power for vertical flight. The nearest equivalent of a dual system for a true helicopter would be to use all the power for a mechanical drive for cruise and part of the power for rotor gas pressure jets during climb and descent.

In such a system, the total h.p. of the engine could be sub-divided according to the duty of the aircraft and the pressure jets would be used for take-off or temporary boost.

The optimum pressure ratio of the pressure jets, if the whole of the lift is obtained from the rotor, is about 4:1. If only part of the lift is obtained the duct passages in the rotor blades are smaller and the optimum pressure ratio, allowing for the power taken from the mechanical drive, is nearer 2:1. This considerably simplifies the auxiliary compressor but the system leans heavily on the design of the rotor, which must now have pressure jets as well as a mechanical drive. It would also need a more complicated engine control. Fig. 6 shows estimated performance of a combined mechanical and gas drive.

6. Boosting of Helicopter Engines

The principal objects of boosting are to increase the take-off load and to meet the engine failure case of a twin-engine machine[14]. By designing the engine to accept boost, it may be made smaller and therefore lighter, to meet a specified emergency requirement.

In the case of the gas turbine applied to the mechanical drive, boosting may be obtained by uprating the gas turbine for a short period. A typical increment is 25 per cent above normal maximum power for two and a half minutes. The turbine is called upon to accept an overload for this period but, as the overload is, or should be, required very rarely, the total period on overload will not affect the engine life, provided the ratings have been thoroughly proved by type test. This manner of boosting is economical in fuel and also tends to improve consumption under cruise, since a lower degree of throttling is used than with an engine made oversize.

With a gas drive, boosting is readily obtained by burning more fuel in the tip jets. A typical normal maximum of 1900°K would permit about another 200°C rise before all the air is burnt.

Although not a turbine system, the rocket-on-rotor has much to commend it for use with a turbine main drive[14]. If such a system using peroxide were generally accepted for helicopters, it would simplify the problem of emergency boost considerably. At present the more active development of the turbo-shaft engine is putting greatest emphasis on meeting all requirements with one power unit. Although the rocket-on-rotor is essentially simple it adds a further system and a new liquid to an already complicated machine.

7. Conclusion

An attempt has been made to discuss briefly some of the design philosophy of gas turbines for helicopters in a general way and to indicate the interesting examples of recent development work in this country.

A small turbo-shaft engine is a specialised form of aircraft gas turbine which, apart from some small but important differences, can expect to suit both the helicopter and the turbo-prop field.

The most popular form of helicopter engine utilises direct mechanical drive to the rotor. Experience with this type verifies that considerable advance is possible in helicopter performance by changing from piston engines to gas turbines.

Gas drives are suitable where rotor lift is more important than endurance. They have a place in small machines where simplicity is essential. The gas drive does away with mechanical transmissions and rotor anti-torque control. Although proved feasible, it has not yet been adopted so widely as mechanical drive.

ACKNOWLEDGMENT

The author wishes to thank members of the staff of D. Napier and Son, Limited who have helped with the preparation of this paper, and also the Managing Director for permission to give the paper. Any opinions expressed are those of the author.

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Fuel Systems For Supersonic Engines

by

E. A. SIMONIS, B.Sc.
(Systems Engineer, de Havilland Engine Co. Ltd.)

Introduction

The advent of the high altitude supersonic aircraft has brought in its train a whole host of complex control problems. As far as the engine manufacture is concerned, these are perhaps best indicated in the diagrammatic layout shown in Fig. 1.

It is not my intention however, to go into the question of multiplicity of controls and their proper co-ordination, nor will I venture into the abstruse province of stability. I should like instead to concentrate on the fuel system itself, *i.e.* the supply and control of fuel to the main engine combustion chamber.

The requirements of the supersonic aircraft, particularly as regards the very wide range of fuel flow rates that have to be covered and the considerable use to which the fuel is needed for cooling purposes, necessitates a revision of existing systems and a reconsideration of the engine control problem from its fundamentals.

Such a study has recently been made within the de Havilland Engine Co. Ltd., and it is proposed now to deal with some of the broader aspects and first, to state the requirements as visualised at the time of the study.

Engine Requirements

Bearing in mind the mixed power plant formula advocated by de Havillands, a flight plan as shown in Fig. 2 has been set down as the requirement for any new supersonic fuel system. If the fuel system is designed and developed from the outset to cover altitudes up to 90,000 ft. and forward speeds up to a Mach

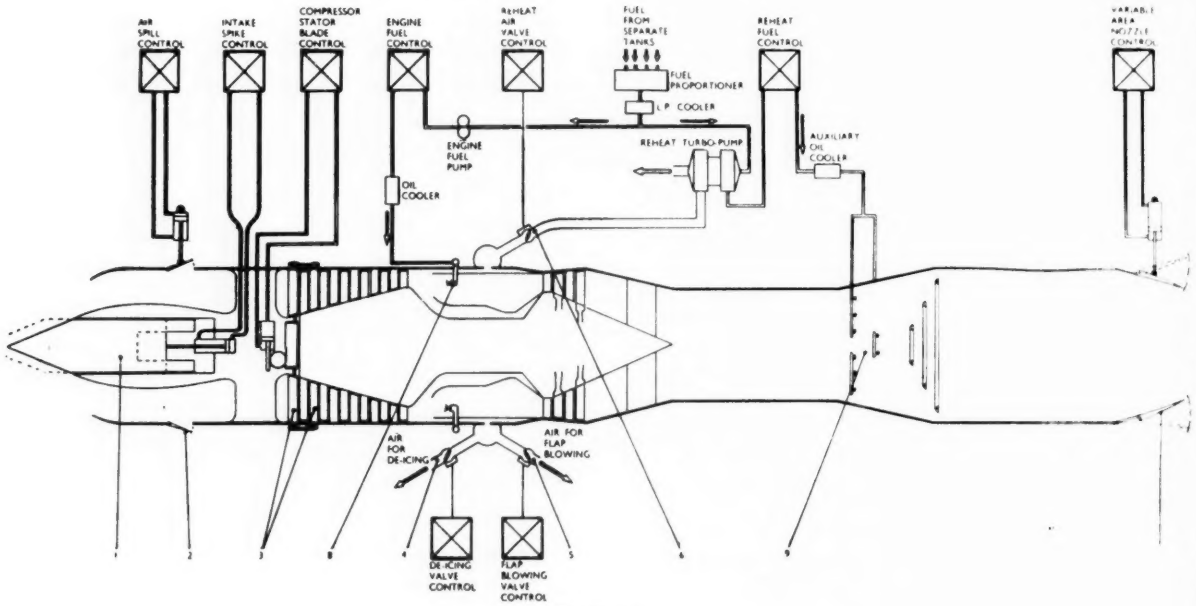


FIGURE 1.

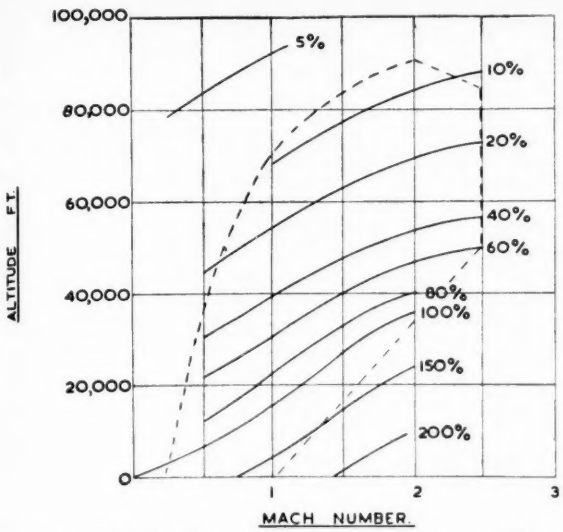


FIGURE 2. Fuel flow for maximum engine output over flight plan.

number of 2·5 then the system should not be the factor limiting the engine or aircraft from attaining these targets.

The curves that have been superimposed on the flight plan are lines of engine full duty fuel consumption expressed as a percentage of the sea level static consumption, *i.e.* the fuel consumption for maximum rotational speed, or maximum turbine temperature, or both. Here we are presented with the first major problem—the large difference in flow rate between high altitude flight and low level high forward speed flight. As shown, the range is of the order of 1:25.

It must be remembered that over the whole flight plan the engine is at, or very near, its maximum safe operating temperature. Therefore control and distribution of fuel must be equally as good at the low rates as at the high rates. Putting these rates into figures; for an engine with a test bed full duty consumption of 1,200 g.p.h., the $M=1$ sea level requirement approaches 1,700 g.p.h. on a normal day, considerably more on a cold day, whereas at 85,000 ft. consumption at $M=2$ is rather less than 100 g.p.h.

Part throttle operation is not expected to alter the range of flow rates to any considerable extent since it is doubtful if the higher altitudes could be attained at anything other than full throttle.

The use of fuel for cooling purposes now seems to be an established fact. The engine designer looks to the fuel as the only means of cooling his engine oil at the high flight speeds envisaged, where ram air is usually above acceptable oil temperatures. This appears a legitimate use and an engine oil cooler can usually be fitted into the high pressure fuel circuit. On the other hand the aircraft designer also covets the cooling properties of the fuel and is incorporating various heat exchangers in the low pressure circuit between the tanks and the engine. Since high speed aircraft draw fuel proportionately from a number of tanks to maintain trim, it is usually unacceptable to return fuel from the cooling circuit to the tanks. Cooling has therefore to be accomplished only with the flow actually being consumed by the engine. At low and medium altitudes the engine consumption rate is adequate: at the higher altitudes the flow is becoming more of a trickle and the fuel temperatures may rise quite substantially through the various coolers, since the cooling requirements are slow in changing.

It is of course essential that the fuel remains liquid at pump inlet and throughout the control system and this brings us to the second major problem. Just where fuel temperatures are starting to rise, fuel pressures are getting low and an adequate margin of system pressure over the fuel vapour pressure has to be maintained if control is to be retained.

Figure 3 indicates the fuel vapour pressure of the three current aviation turbine fuels AVCAT, AVTUR and AVTAG (wide-cut). This is information based on recent experiments by the Shell Company. It shows how very sharply the vapour pressure rises as the fuel temperature increases and some margin above these curves has to be maintained to ensure no local gas release areas.

Although it would be convenient to stipulate that all supersonic aircraft should operate on AVCAT, or even on a light diesel oil and thus dodge this issue, we have insisted that the system should be capable of operating the flight plan on AVTAG. Any easement would be welcome, but again if the system is designed from the outset to operate on wide-cut gasoline it should have some margin in hand when on kerosine.

The use of fuel at high temperatures in supersonic engine control systems brings the added hazard of gum or lacquer deposition in burners, coolers, and so on, or ultimately, even to fuel cracking. In the United States this has assumed sufficient importance for a modification to the general fuel specification to be introduced.

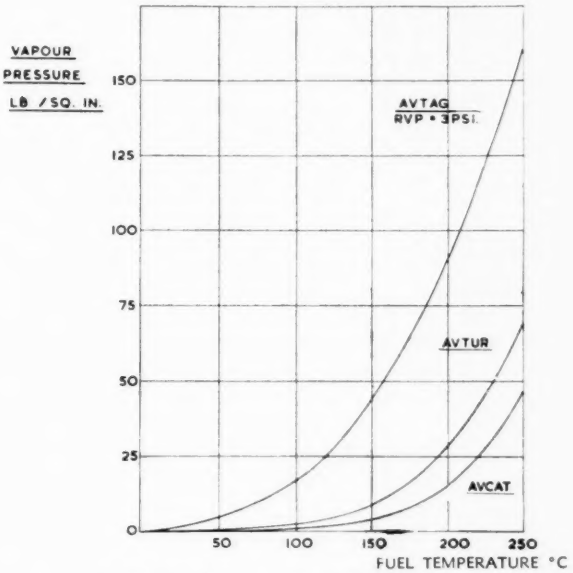


FIGURE 3. Vapour pressure of gas turbine fuels
(From information supplied by Shell.)

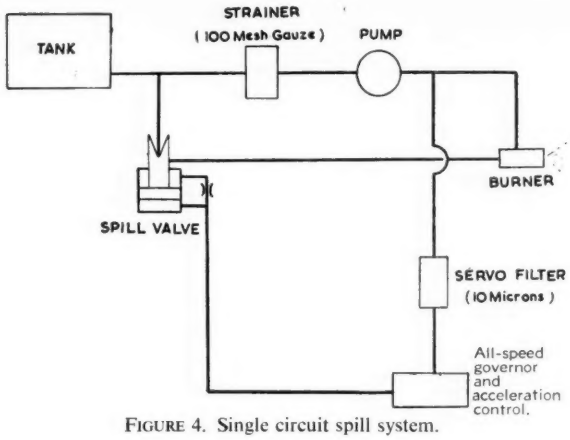


FIGURE 4. Single circuit spill system.

All fuels will shortly have to pass a test that has become known as the ERDCO test. This test assesses on a qualitative, rather than a quantitative, basis the gum and lacquer deposition tendencies of the fuel at high temperatures.

One other requirement has to be borne in mind, that relating to dirty fuels. The number of engine failures in the past attributable to small particles of dirt upsetting the fuel control system form quite a high proportion of the total, and it has long been the aim of all engineers associated with this side of the business to render their pumps, control units and burners less sensitive to dirt. The target set has been for the main fuel flow to be filtered solely by a fine mesh gauze—about 150 mesh is under consideration—and to restrict any finer filtration to the small quantities of flow needed for servo control; *i.e.* pumps, throttles and burners must pass or absorb without detriment particles of dirt up to about 80 microns in size but servo valves will retain protection against particles of approximately 6 microns, as has been previous practice.

The ability to withstand ice is a requirement associated more with subsonic than with supersonic aircraft. The freezing of water suspensions in fuel tanks or pipe runs, with a consequent clogging of filters, control units or even burners is a problem that has been given a great deal of attention over the past years, as the results can be so sudden and unwelcome. The time element, however, enters into this problem to quite a considerable extent and it is doubtful if supersonic aircraft will be long enough in the air for this danger to arise, or else they will be flying so fast that overheating, rather than freezing, will be the problem.

Summarising, the requirements become mainly those associated with the uniform injection of fuel at rates varying by as much as 25 to one with increasing temperature as the rate drops. Let us now investigate how this can be done.

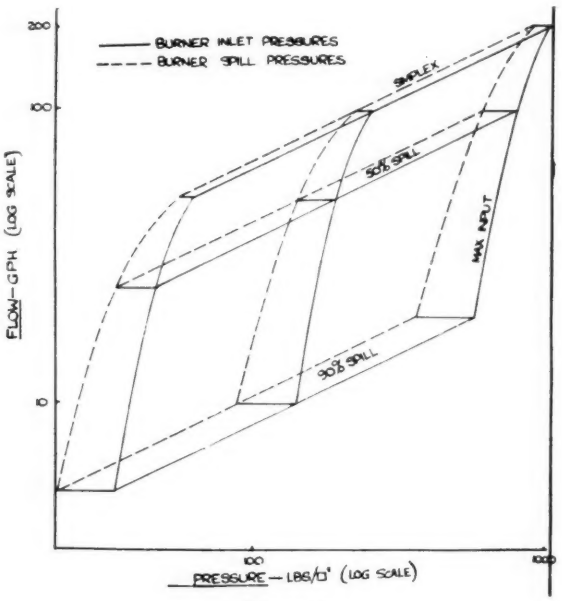


FIGURE 5.

Injection Methods

Taking first the simple direct injection burners with a flow proportional to the square root of the pressure drop across them, a 25 to one flow rate variation implies a 625 to one pressure drop change. If the pressure at the lowest flow rate is to be sufficient to prevent boiling, then the pressure at the highest rate becomes quite prohibitive.

The need for duplex or even triplex burners becomes apparent if direct injection is to be used. This implies

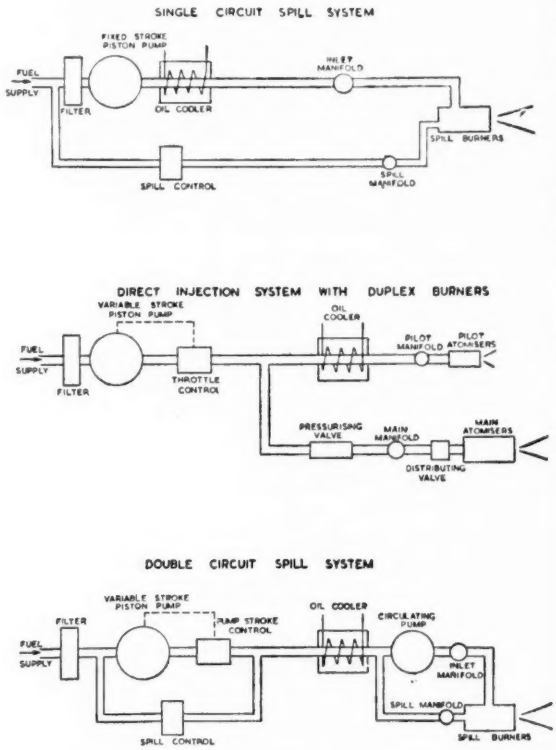


FIGURE 6.

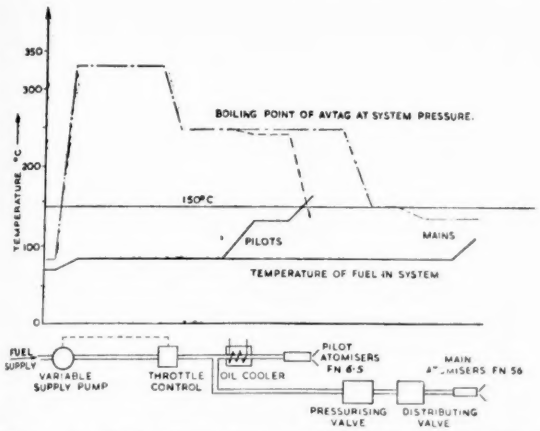


FIGURE 7. Fuel temperatures at 60,000 ft. $M=2$. Direct injection system with duplex burners.

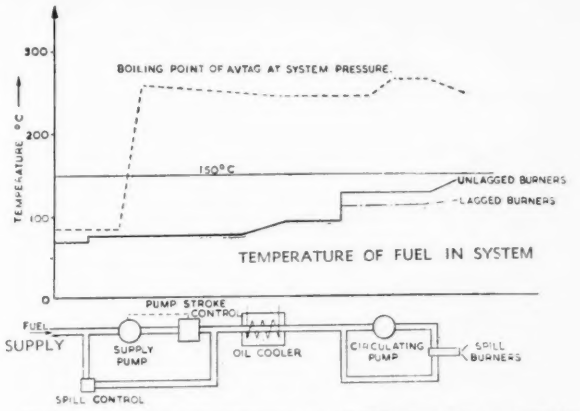


FIGURE 8. Fuel temperatures at 60,000 ft. $M=2$. Double circuit spill system.

sets of burners or atomisers being brought into use progressively as the flow rate increases and involves pressurising or change-over valves. A serious distribution problem exists at the change-over point where the manifold pressure to the burners just being opened up is low and can be comparable to the pressure head difference between upper and lower burners on an engine. Without distributors of some sort the flow to the bottom portion of the engine must then be somewhat greater than to the top portion, which will lead to an uneven temperature pattern around the turbine with the possibility of distortion and damage.

Distributors for low flow rates are difficult to manufacture and de Havillands have had little success with them.

The alternative to direct injection is the spill burner. A simple single circuit spill burner system as shown in Fig. 4 offers many advantages for subsonic application. It is, however, unsuited for the high altitude supersonic engine on account of the return of hot fuel to the low pressure side of the pump. With a fixed stroke pump and at low engine fuel requirements considerable pumping energy is wasted. Most of this is converted into heat and is circulated around the spill path. Heat is also picked up by the spill fuel in its passage into and out of the combustion chambers and, athough this can be to some extent obviated by lagging the burner stem, high spill temperatures are unavoidable.

It becomes necessary if a spill burner system is to be utilised to make two modifications to the simple circuit shown:—

(*i*) to replace the fixed stroke pump by one of varying stroke so that the pumping energy can be reduced at altitude;

and (*ii*) to return the hot spill flow to a high pressure part of the system where vaporisation can be avoided.

A typical spill burner characteristic is shown in Fig. 5. The more vertical lines represent flow-pressure relationships at different circulating rates. Taking the highest circulating rate, it will be observed that the desired 25:1 flow rate can be obtained with an inlet pressure change from 1,000 p.s.i. to between 400 and 500 p.s.i. and a spill pressure change from 780 p.s.i. to something well over 100 p.s.i. The pressures in every

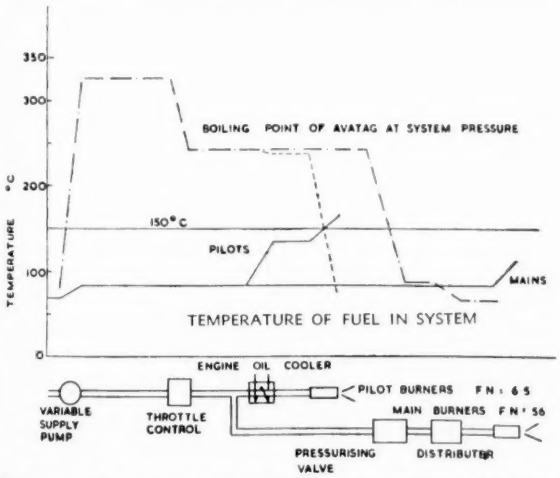


FIGURE 9. Fuel temperature at 85,000 ft. $M=2$. Direct injection system with duplex burners.

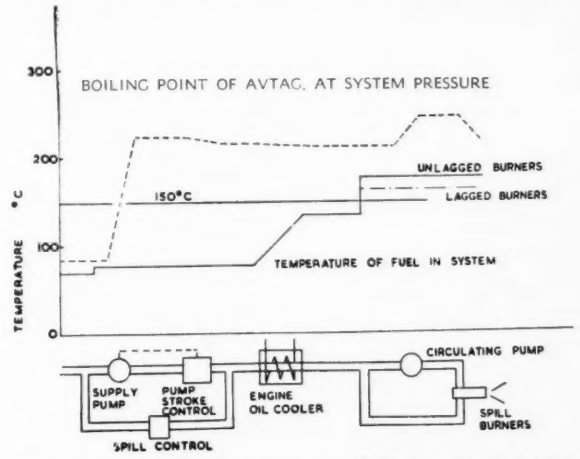


FIGURE 10. Fuel temperature at 85,000 ft. $M=2$. Double circuit spill system.

case are well above the head difference, which consequently can have negligible effect on distribution.

The constant circulating rate for a given engine speed can be obtained from an engine-driven centrifugal pump which generates a pressure rise proportional to speed squared. This pressure rise will then be substantially constant for a given engine speed and, if placed across the constant flow orifices of the spill burners, will maintain reasonably constant rate.

This introduces the concept of the two circuit spill system shown in the lower diagram of Fig. 6. The supply pump, control unit and spill valve form one circuit; the burners and circulating pump the other. It is convenient to put the engine oil cooler in the flow line between the circuits.

Although the flow rate around the burner circuit is now held constant at a given engine speed, as is the pressure rise across the circulating pump or drop across the burners, the actual level of pressure depends on the flow into the circuit and out of the burners. The high temperature spill flow now returns after the oil cooler and at a substantial pressure.

A fairly detailed investigation was made to determine whether this system had in fact any theoretical advantages over the more generally accepted system shown in the middle diagram as regards the margin from fuel vaporisation troubles. The results of this investigation are given in Figures 7-10.

The first two (Figs. 7 and 8), consider conditions at 60,000 ft. and a Mach number of 2·0.

I have plotted fuel temperature through the system assuming it arrives into the fuel pump from the aircraft coolers at 70°C. A constant heat input from the engine oil cooler is also assumed. Shown dotted is the boiling point of AVTAG at the fuel pressures arising in the various parts of the system.

It will be seen that both direct injection and spill systems have an adequate margin throughout, though the margin from boiling is somewhat greater for the spill system.

At 85,000 ft., and a Mach number of 2, things are a little different. (Figs. 9 and 10). With direct injection burners, vaporisation will occur in the main flow line after the pressurising valve and will probably also occur in the pilot atomisers themselves. If the control unit is of the flow metering type, rather than of the pressure control type, vaporisation will result in an increase in pressure to pass the metered quantity as a gas instead of a liquid. The increase in pressure will itself tend to suppress boiling and some compensation is inherent. The danger, however, is that vaporisation will occur locally at a few burners only and distribution will be badly affected. It must also be reckoned that at lower flows still, the flow into the main line will cease, leaving a stagnant quantity of fuel at high temperature to evaporate off. There is then a considerable possibility of gum or lacquer being deposited on the distributor or burner parts.

Theoretically the spill system still has a margin at 85,000 ft. and a Mach number of 2. Although the fuel temperatures are higher than with direct injection, so are the pressures. The circulating flow rate is also high so that any deposits from the fuel can be expected to be washed away and, in any case, should not clog the relatively large orifices of the burners.

Installation

In the past, control systems have tended to evolve as the engine developed, so that additional functions were added as the need arose. These tended to become separate control units which were piped into the main control as best suited the installation at the time. As the years passed so the engine looked more and more like a Christmas tree, with control units festooned around it and pipes running in all directions.

What started off as evolution and could at the time be justified, has tended to become a habit. The weakness of pipes and their joints and the tendency to pick up stray vibrations, provides quite a serious hazard to aircraft safety. Furthermore, every time a pipe joint is broken there is the danger of a speck of dirt entering the control system downstream of the filter.

The integration of all fuel controls into a common assembly thus becomes a definite part of a policy to render the engine less sensitive to minor troubles and more suitable for service under field conditions.

Exhaust Nozzles For Supersonic Aircraft

by

H. PEARSON, B.A., F.R.Ae.S.
(Chief Research Engineer, Rolls-Royce Ltd.)

THE ORDINARY convergent exhaust nozzle is a perfectly satisfactory and efficient device for converting pressure in the jet pipe into velocity, so long as the pressure ratio across it remains below about 2·0; or, put another way, so long as the velocity at exit from the nozzle is not theoretically greater than that of sound at the jet temperature prevailing. If the pressure ratio is greater than this however, full expansion cannot take place and some losses in thrust occur. The pressure ratio across a jet pipe on modern jet engines is of the order of 2½ on the test bed, so already under these conditions some losses occur. The amount is insignificant however, but in flight at supersonic speeds the pressure ratio across the jet pipe may become very large, and in this case important losses result.

For instance, a typical supersonic engine flying in

the stratosphere at a Mach number of 2·5 will have an expansion ratio across the jet nozzle of between 15 and 20. If the exhaust gases could be fully expanded satisfactorily, then the jet pipe thrust would be increased by nearly 12 per cent at this condition. Fig. 1 shows the increase in gross thrust obtainable by efficient expansion of the exhaust plotted against flight Mach number. The effect is becoming marked when aircraft speeds exceed that of sound.

Now the actual thrust produced by the engine is the difference between the gross thrust and the intake drag. At high speeds both these quantities are becoming large and thus the improvement as a percentage of the net thrust is much more marked, as seen on Fig. 2. It will be seen that at the speed of sound about 5 per cent increase in net thrust can be obtained, about 12 per cent at a Mach number of 1·5, about 24 per cent at a Mach number of 2·0 and 45 per cent at a Mach number of 2·5. These are very large gains indeed.

Convergent-Divergent Nozzles

One method of expanding the exhaust efficiently when the pressure ratio is high is to use a convergent-divergent nozzle, which is nothing more than adding a divergent section to the back end of a convergent nozzle. In this way the exhaust gases expanding after the throat can exert a reaction on the divergent walls, and thus more thrust can be obtained. Fig. 3 illustrates roughly what happens in the two types of nozzle. In the convergent nozzle the expansion proceeds, but there is nothing to prevent sideways expansion and in any case nothing for the expanding gases to push on. In the convergent-divergent nozzle this sideways expansion is restricted and the divergent portion provides a rearward facing area for the gases to thrust on.

The gains from fitting a convergent-divergent nozzle are to some extent offset by some losses due to friction but, in general, it is possible to achieve the theoretical gross thrust within, say, 2 per cent at the high pressure ratios. However, there are important limitations to the use of such a nozzle. First, to expand the gases efficiently under any conditions the area of the outlet of the divergent portion must bear a definite relation to the area of the throat. At very high aircraft speeds the outlet area required is so large that the nozzle would be much larger than the engine and would cause more drag. Thus some compromise on area ratio has to be made and an area ratio of around 1·7-1·8 is usually taken as the limit.

Secondly, for a nozzle of such a fixed ratio, when the aircraft is at low speeds the area ratio is too high, shock losses occur and there is a large loss in thrust. Fig. 4 shows the net thrust increase plotted against

FIGURE 3. Supersonic expansion from convergent and within convergent-divergent nozzles.

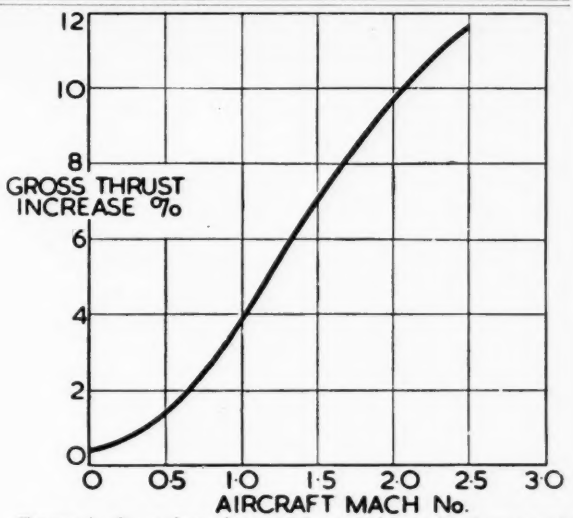


FIGURE 1. Gross thrust increase due to using an ideal supersonic nozzle instead of a convergent nozzle on a typical supersonic engine.

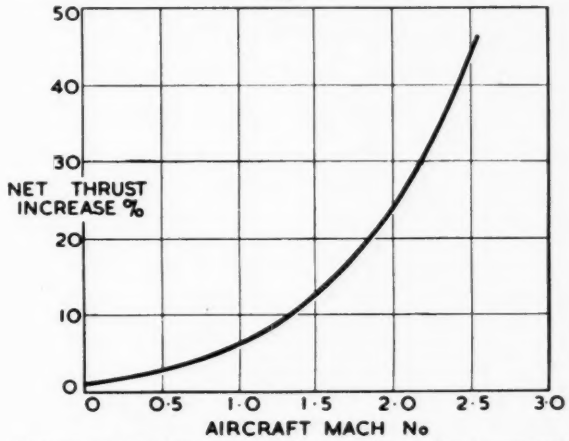


FIGURE 2. Net thrust increase due to using an ideal supersonic nozzle instead of a convergent nozzle on a typical supersonic engine.

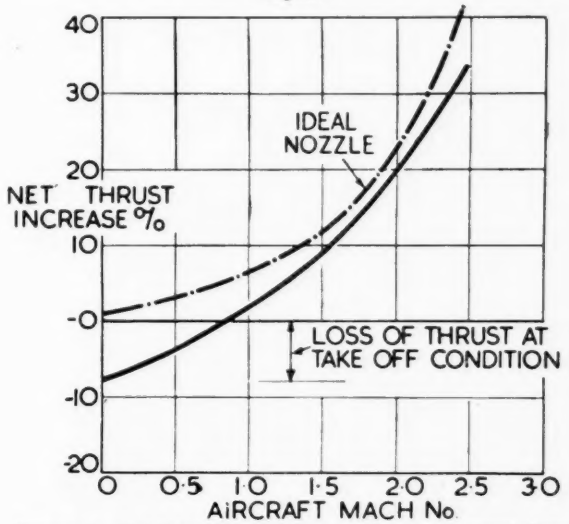


FIGURE 4. Net thrust increase due to using a convergent-divergent nozzle with fixed area ratio of 1·73.

aircraft Mach number for a typical engine, assuming a convergent-divergent nozzle of area ratio 1·73, compared with the theoretical line previously shown. There is a small loss at around $M=1·5$ due to the frictional losses of the nozzle, a larger reduction at the highest flight speeds, due to the restriction on the area ratio previously described; and quite a serious loss at very low speeds and take-off due to the area ratio being too large for these conditions.

It is evident that a fixed convergent-divergent nozzle is not entirely a satisfactory device unless only moderate supersonic speeds are to be considered. There are two further limitations to its use; first, at low aircraft speeds there is not only a serious loss in thrust but violent instability of the flow may be set up, causing probable rapid failure of the jet pipe or engine. Secondly, if reheat is necessary, as on a fighter aircraft, the convergent-divergent nozzle must in any case be variable so a fixed convergent-divergent nozzle is hardly possible.

Variable Convergent-Divergent Nozzles

An obvious way to overcome the previous disadvantages is to make the nozzle variable. In the general case this means adding a variable divergent portion to an already variable throat area convergent nozzle. Mechanically, this is a difficult job and in addition, operating rams and control systems are necessary. We have nevertheless explored the possibility of such nozzles and have made and tested experimental nozzles. Fig. 5 shows such a nozzle as tested on an actual engine, but this is purely an experimental set-up as regards the method of controlling and operating.

There are important additional losses in such a nozzle due to imperfect shape and sealing. Fig. 6 shows the performance of such a nozzle compared with that of the fixed nozzle. The loss at low speeds has been considerably reduced, although not completely eliminated, while there is some additional loss at the higher aircraft speeds due to the extra losses mentioned.

Such a nozzle would avoid instability at low aircraft speeds and is capable of giving about 27 per cent

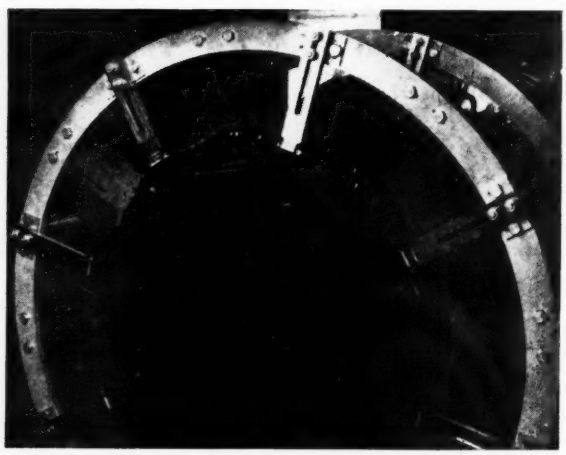


FIGURE 5.

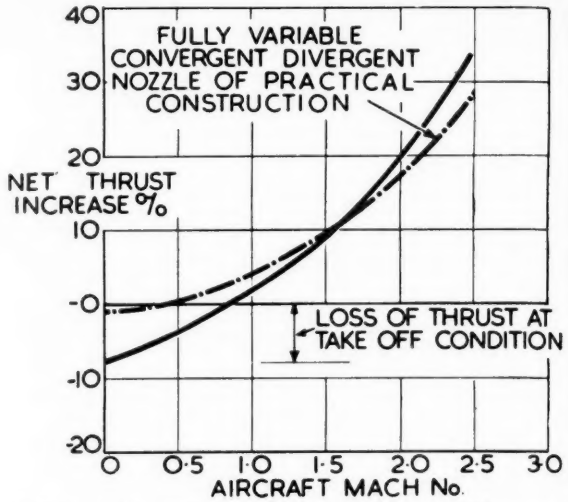


FIGURE 6. Net thrust increase due to using a variable convergent-divergent nozzle.

increase in thrust at $M=2·5$. It would, however, be heavy and complicated; nevertheless, for supersonic aircraft spending appreciable portions of their endurance at high speeds it would be a worth-while device in spite of these disadvantages.

Ejector Nozzles

It is always enticing to attempt to use aerodynamic methods of control when mechanical methods become cumbersome. In America it has always been the practice to cool jet pipes with relatively large amounts of air bled from the intake and if this cooling air could be made to control the effective area ratio of a convergent-divergent nozzle, a simple system might be achieved. In this way the ejector supersonic nozzle was evolved and Fig. 7 shows diagrammatically such a nozzle.

It consists essentially of a normal convergent primary nozzle, which in the general case may be made variable if required for reheat, and in all respects the engine and jet pipe are as normal. The primary nozzle is surrounded by a cylindrical shroud of approximately the dimensions indicated, the shape being somewhat a matter of compromise, and fed by a source of air which may be either the main intake or a subsidiary intake. Now if the issuing primary jet has a pressure ratio greater than 2·0 it will attempt to expand after leaving the nozzle. The amount of this may be controlled by the pressure of the surrounding air, which at

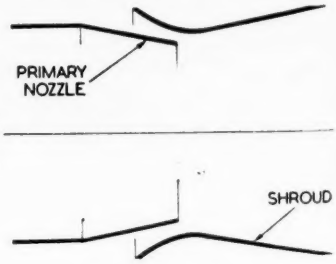


FIGURE 7. Principles of an ejector nozzle.

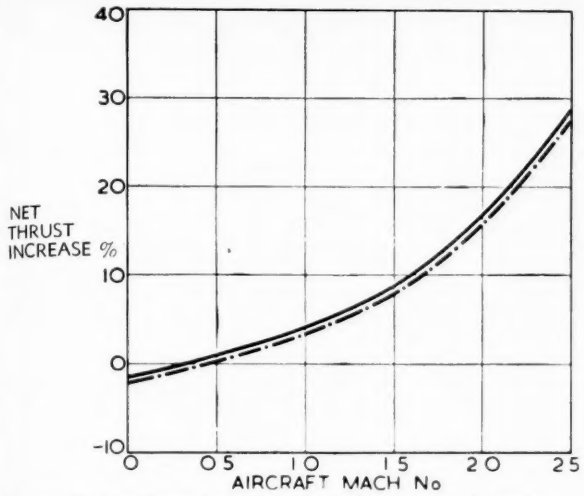


FIGURE 8. Net thrust increase of a typical supersonic engine. Replacing the convergent nozzle by:
(*i*) Mechanically variable nozzle ————
(*ii*) An ejector nozzle — - — - —

high supersonic speeds is made relatively low so that the expanding supersonic jet may nearly fill the shroud space. At low speeds the secondary air pressure is made relatively high, thus preventing any over-expansion of the primary jet and so internal losses and instability are avoided.

Obviously there are some losses in such an arrangement; but such nozzles are capable of producing almost as much thrust as a mechanically variable convergent-divergent nozzle, as shown in Fig. 8. To arrive at this comparison it is, of course, necessary, not only to allow for the thrust from primary and secondary air flows but, also, to deduct the inlet momentum or ram drag of the secondary air.

The requirements of secondary air for such a nozzle are shown in Fig. 9. It will be seen that at low speeds more than 10 per cent of secondary air must be passed,

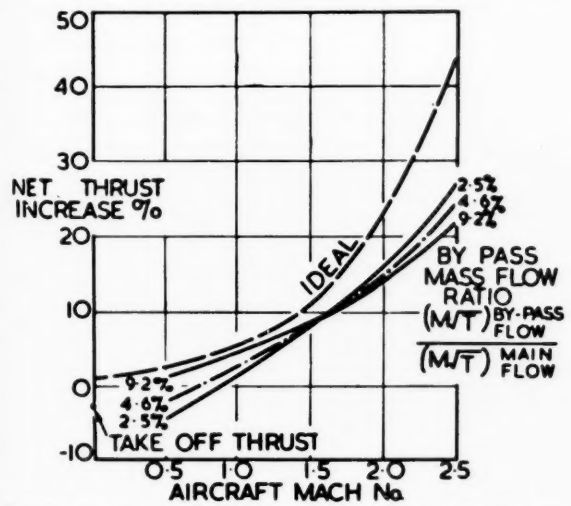


FIGURE 9. Net thrust increase of a typical supersonic engine due to replacing the convergent nozzle by an ejector nozzle.

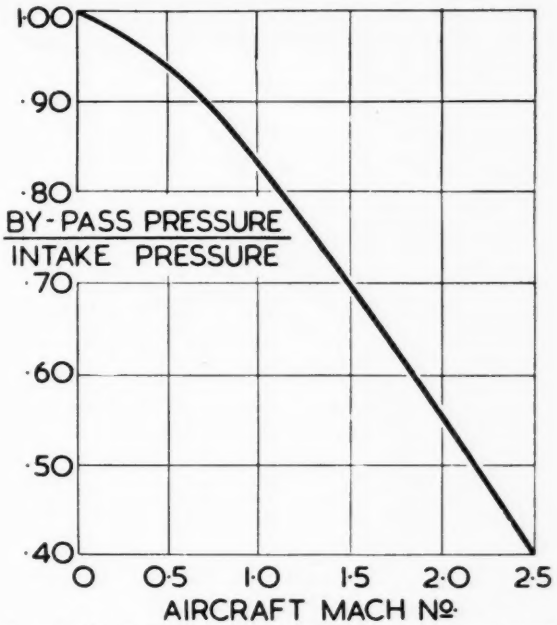


FIGURE 10. By-pass pressure required by an ejector nozzle applied to a typical supersonic engine.

while at high supersonic speeds the secondary air flow must be reduced virtually to zero. In practice a minimum amount sufficient to keep the shroud reasonably cool is used. To achieve these flows requires pressures in the secondary air, as shown in Fig. 10. At low speeds pressures substantially equal to that given by the main air intake are required, but at high speeds the pressure required is much lower. This means that if the secondary air is supplied from the main air intake it has to be substantially throttled at high speeds, this throttling being removed progressively at lower speeds.

Figure 11 shows a typical installation diagrammatically. The main air intake duct to the engine has spring-controlled doors which lead to a duct feeding the rear jet pipe region and secondary nozzle. A valve in this duct is either controlled manually or automatically to regulate the secondary flow to the optimum quantity. The air is ducted through to the bulkhead, thus keeping the main fire zones apart. At take-off there will be a pressure below atmospheric in the main intake duct and to avoid the possibility of exhaust being drawn backwards through the ducts into the intake, the

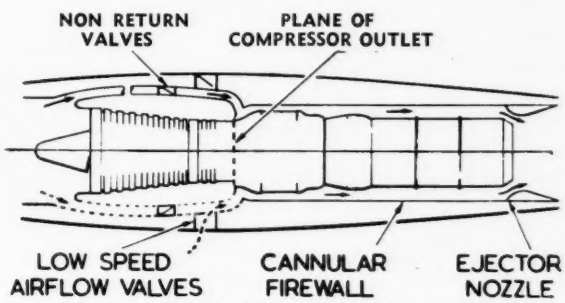


FIGURE 11. Typical installation of ejector nozzle.

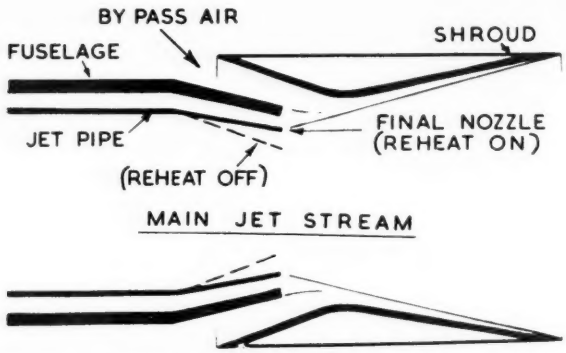


FIGURE 12. Diagram of ejector nozzle.

flap valves on the intake automaticaly close and an auxiliary intake in the fuselage opens up.

In tightly fitting installations there is often no room for air to be passed round the engine. For such a case there exists the possibility of feeding the secondary air to the ejector nozzle by means of a rear intake, as shown in Fig. 12. Such an intake should be reasonably efficient at low speeds but poor at supersonic speeds, so that there is some hope that an intake suitably developed would have just the right flow and pressure characteristics required by the nozzle, without any adjustment being necessary. Care has to be exercised that this rear air intake does not cause any large increase in aircraft drag.

Matching of Air Intakes and Nozzles

As shown previously, the ejector nozzle is just another way of providing a variable convergent-divergent nozzle without the weight and complexity of a mechanically variable nozzle. Where the secondary air is fed from the main air intake, suitable matching of the air intake has to be considered.

In a supersonic aircraft fitted with a fixed air intake it is invariably found that, if the intake size is chosen to suit the engine requirements at, say, a Mach number of 1·0, then at around $M=2{\cdot}0$ the air intake size is much too big, with the result that air is spilled round the intake and a large "spillage" drag is set up. Conversely, at lower speeds the air intake size is not really big enough and an auxiliary intake, only open at low speeds, has to be provided. Thus a fixed air intake has air to spare at high Mach numbers and a deficit at low speeds. This is just the inverse of what is required by the ejector nozzle and at first sight it would not seem possible to feed it from the main intake.

This difficulty has been overcome by making the secondary air flow shroud variable, as shown in Fig. 13. At low speeds the shroud is closed to the small area position and a minimum quantity of air only sufficient to cool it is passed. At high aircraft speeds the shroud is opened up roughly to the parallel position, when the area ratio is large. A large quantity of air from the main air intake is then fed to the secondary, with the result that full supersonic jet expansion plus reduced spillage drag is obtained. A variable nozzle to provide reheat may also be incorporated.

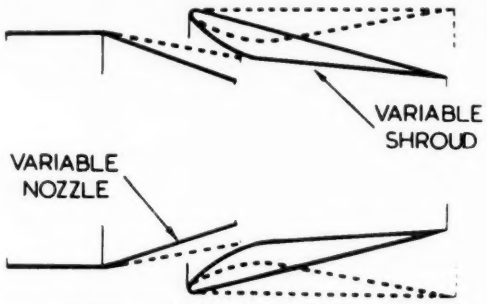


FIGURE 13. Variable shroud ejector nozzle.

It might seem that in thus making a variable shroud, the complexity of a variable nozzle has been reintroduced, but the mechanical problem of providing such a variable shroud would be much less than for a variable main jet nozzle.

The ideal air intake must be made variable if good intake efficiency over a wide speed range is required. If, for instance, a two-dimensional air intake is made variable by varying the cone angle of the wedge, then it is found that the air flow characteristics are just the reverse of those for the fixed intake. This means that if the area is chosen to match the engine requirements at high Mach number, then the air intake will provide an excess flow at low Mach numbers. This is what is required by a fixed ejector nozzle, and so the ideal combination of a variable air intake and a fixed ejector shroud may be the recommended arrangement for very high speed aircraft.

Obviously such a conclusion is subject to all the various engineering considerations involved in a complicated engine installation. In addition, it is necessary to consider the effect of the external flow round the back end of the aircraft and its reaction on the jet stream. Such studies are being made, but their discussion is outside the scope of a necessarily restricted paper such as this.

Conclusions

(1) Theoretically there is a large thrust gain available by efficiently expanding exhaust gases at high aircraft speeds. Compared with the normal convergent nozzle, at an aircraft Mach number of 2·5 there is theoretically 45 per cent thrust increase possible.

(2) By practical forms of nozzle a thrust increase at $M=2{\cdot}5$ of about 30 per cent is achievable on known test data.

(3) To avoid serious off-design losses a mechanically variable convergent-divergent nozzle or an ejector nozzle are required. There is little to choose on exhaust performance between the two, but the latter is probably much lighter and less difficult mechanically.

(4) The ejector nozzle, when used in conjunction with a fixed main air intake has to have a variable shroud. When used in conjunction with a variable air intake it can be fixed and the two form an ideal combination of high thrust and low drag.

Problems in the Development of a Guided Weapon*

by

J. CLEMOW, M.A.

(Chief Engineer (Weapons), Vickers-Armstrongs (Aircraft) Ltd.)

1. Introduction

The subject of this paper is a wide one. Guided weapon development problems in one form or another have been my daily concern for more than ten years now and I must be highly selective in the brief time at my disposal on this occasion.

The object of guided missile development is to produce efficient weapons of war, that is, weapons of destruction able to neutralise an enemy threat or destroy the enemy if he dare attack. I make this statement every time I talk on guided weapons, and I make no apology for making it on this occasion. We must not at any time forget that we are concerned in the development of weapons to defend this country, that time is of the essence of the contract, that the product must stand up to storage, to usage by the Services, and must be ready for action after little warning, and so on. As in any other branch of engineering, we have to make the best use of the men, ideas, techniques and materials at our disposal in order to reach the desired objective.

2. Organisation

By and large I have found little to complain about in the properties of materials or the laws of nature, although sometimes aggravating, they are generally pretty consistent. I would say that most of the problems have arisen in how to get the best out of the team of men at one's disposal. A great deal has been said about man management and most of it could be repeated again with profit. But I am not so much thinking here of the general principles which we all know and all too often forget to apply, such as, for example, appreciating that when we hire a man we hire his bad points as well as his good, and it is for us to make the best of his good qualities and arrange that his bad points are at least neutralised; also the principle of delegating responsibility and so on. What I have in mind here is that in guided weapon development we have many different branches of engineering brought into close proximity, the success of the whole guided weapon system depending on the successful integration of these different fields.

It is all too easy to fall into the habit of thinking that the field in which one is particularly expert is the most important one in the project. Not only is it important to get away from this quite erroneous attitude, it is important to go further and make positive steps, certainly if you are occupying any senior position in a guided weapon team, to understand the problems and method of thinking of the other fields of science and technology which are concerned in the weapon system. Designers, laboratory engineers, the work people who make the components, the men who assemble the rounds and conduct the trials, and so on, all are equally important to the project; every one of them is just as likely to make the mistake which ruins the trial and the success of the enterprise.

I find that about four subordinates is the number that I like to have reporting directly to me. Certainly I would draw the line at six. This is fortunate because I have found that guided weapon work can be broken down into three technical divisions and one administrative one. One can regard these first of all as a Design Division, secondly an Experimental Workshop Division, thirdly a Test and Trials Division, and lastly, but not least important, an Administrative Division which looks after the important matters of pay, allowances, contracts, keeping the books in order and so on. I regard a good Administrative Division, tactfully and helpfully run, as an essential part of the organisation in that, if it is good, it can do a great deal to relieve the technical staff of rather irritating everyday chores and provide the satisfactory environment in which they can work. You will all know of the score of things that tend to interrupt the daily work of the scientist or engineer, and will not need any amplification on this.

Each of these Divisions again can be conveniently broken down into about the same number, that is to say four or five, branches of activity. One can, for example, break the Design Division down into the following branches: Aerodynamics and Technical Office, Structure and Propulsion, Electronics, Auto-pilot and Instrumentation. The Works Division, in development work at any rate, seems to me to break down into a Planning Branch, a Model Shop or Prototype Shop, and an Experimental Works. The distinction that I draw here is that the Model Shop or Prototype Shop, call it what you will, makes the models for laboratory use and for general purposes, but only in exceptional cases makes anything which goes into a round to be fired. I have tried to make it a rule that anything which goes into a round fired on a trial should come from the main Experimental Works where it has been subjected to

*The fourth lecture to be given before the Guided Flight Section of the Society—on 20th March 1958.

proper drawings, modification procedure, inspection, and so on. Frankly I think it rather vain to talk about reliability until the components come from some disciplined organisation. The Trials Division breaks down, again I think conveniently, into a Preparation and Test Branch, a Trials Branch actually responsible for the conduct of the trial, and an Analysis Branch. One would probably also have here a Test Gear Branch and, if the missile is one that is fired from aircraft, it is probably convenient to have a Branch responsible for the preparation of the aircraft.

Having mentioned test gear, I would like to say just a little about it. Firstly, the test gear is an important part of the weapon system. It is just as important that it should be designed for Service use as that the rest of the weapon should be. It is for this reason that I have a preference for putting the design and engineering of the test equipment in the Trials Group as their responsibility; for it is these people who have the experience of carrying out trials with development missiles, under often very difficult conditions, at the ranges. None should know better than they what can best be done to make life easier in this area. On the other hand, of course, the members of the Trials Department will not have such an intimate knowledge of the workings of the system as the original designer. Therefore, the system which I try to work is the following.

The designer of any component of the missile, say a gyroscope, is responsible for designing and developing not only his gyroscope but the accompanying test gear that will test this gyroscope. During development test the test gear evolves along with the component, and this particular unit of test gear, say for the gyroscope, then serves as the start for two different purposes. First there is the test gear that will be required by the Experimental Works, and indeed later by the production line, and secondly there is the test gear, overall test gear if you like, that will be required by the Trials team and later by the Service unit to set up and check the completed missile. Thus the Trials Division takes the basic bricks of test gear as designed and to some extent developed by the Design Division, and engineers them into suitable form for trials and later for Service purposes. Similarly, of course, the Workshops must see that test gear suitable for their purposes is designed from the original Design Division equipment.

There are two features about the organisation which I have outlined that you may think surprising. The first is the absence of any group labelled "New Projects Department" or "Assessment Department," and the next is the absence of any area called "Research Branch." Dealing with the second point first, I firmly believe that the head of any branch, let us say the Electronics Branch, must be the head, not only of the work currently in hand, but must also be responsible for any forward-looking work that is going on. In my experience I have never found it successful to separate a group of men and tell them to get on with forward-looking work, forward-looking investigations, for any length of time in isolation from their engineers engaged in the current work. All too soon they get out of touch with realities and will have both feet off the ground. It is for this same reason that I am not particularly keen to have a branch specially labelled "New Projects." I prefer to form up an Assessment team *ad hoc* by choosing one man, perhaps on some occasions two men, from each of the branches concerned, and forming them together with one Project Leader or Project Officer to carry out the assessment of the particular weapon.

It is convenient to use one branch as a sort of filing centre or editorial office, and I find it convenient from the very nature of their work on performance calculations and so on to use the Aerodynamics Branch for this. By forming up an Assessment team in this way from engineers actively engaged on current work I find that I get a practical outlook on the problem in hand, and also avoid the difficulty that sometimes arises when a separate, divorced Assessment Group is used, that when the work later reaches the Design Department it is not liked particularly and in some measure gets torn up. But also there is great educational benefit in conducting the assessment work in this way. It gives the specialist engineer an opportunity to work in close contact with his colleagues in the other branches. He has to learn the give and take of engineering a rather complicated set of techniques, which the guided weapon is, he gets a good insight into the interactions between one part of the equipment and another, he is brought in contact with the more senior members of the team when he has to write a concise and intelligent report on his part of the project and he is similarly brought into contact with the senior members of associated firms. Briefly he is taken outside the four walls of his laboratory.

Usually a firm's Guided Weapon Department will be handling a number of guided weapon projects in different stages of development. The question arises as to how to handle these, *i.e.* whether to break one's organisation up project-wise or by functions. You have gathered from what I have already said that my preference is for an organisation which is based on function. Frankly I would hate to have to operate a system where there were two electronics departments, one working on Project A and one working on Project B. As far as I am concerned, there is only one Electronics Branch in a Guided Weapon Department. I prefer to carry out the control of a number of projects by the Project Officer system, as I indicated when talking about assessment.

The Project Officer for a given project (and Project Officers have to be quite remarkable men in their breadth of vision and qualities of leadership and personality) must have no interest other than seeing his project through. I visualise his control as running horizontally across all the Design, Works and Trials Branches of the organisation. His is the responsibility for seeing that the project keeps to its development programme. While the head of a branch has complete sovereignty over his own area, it is for the Project Officer, who should be a man of the same stature, to represent to any Group Leader that not enough effort is being put on this or that aspect, to point out that to him there is at least a *prima facie* case that the design or

manufacture is not satisfactory in this or that respect, and so on.

Now for the top end of the organisation. Titles vary throughout the Industry: I am called "Chief Engineer," so for purposes of argument we will use that title. The system to which I have always worked is to have a deputy who, as it were, runs in double harness to myself and who is as far as possible as well aware of the situation in all fields as I am. There is a little more to it than that. I normally try to break down the work so that there are reporting to me first of all my deputy, who is responsible for all the lines of control vertically, that is to say the heads of the Design, Works and Trials Divisions all report to him; one might call this the internal running of the Department. Additionally I have reporting to me the Project Officers whose lines of control, as I have said earlier, run horizontally across the system. You will see then, in the familiar language of control engineers, that we have got a closed loop system.

Right from the design study stage I insist on a fairly detailed programme. It pays to break this down into considerable detail and get each Branch Leader to forecast how long and how many men it will take to conduct, say, the design, development and testing of an actuator. He may in the event turn out to be wildly wrong, but never mind, he has had to sit down and think about it, and there is some yardstick by which to measure. Again I insist on a design data book, or design compendium, being kept from the earliest stages. Otherwise I find it is all too easy to go adrift and to base your design on some inconsistency. It is only too easy to have two or three, or even more, sets of aerodynamic derivatives available around the place, and the same goes for other parts of the system.

The firing programme wants particular care. It is not the slightest good firing a missile if its components have not behaved satisfactorily on the ground when subjected to environmental tests of one sort or another. There is no point at all in firing rounds to prove something that you could quite adequately demonstrate by static firings on the ground. This we all know full well, but it is surprising how woolly some of the objects of some trials can be. It is not very much use writing on one's programme: "Six firings to prove the control system." Some thought must be paid to what you really want to get out of the firings and the trials planned with that in mind.

Here I must make mention of the Law of Pure Cussedness, in which I most firmly believe. Out of my respect for this law I endeavour mostly to fire rounds in pairs, allegedly identical, under conditions which are as identical as the weather at Aberporth or Larkhill will permit. It is surprising how different the records from two successful and allegedly identical rounds can look.

3. Trials

The only way to conduct trials is to know definitely what you are seeking to find out from the particular trial; to have formed a fairly good opinion from theoretical work or simulator runs and the like, of what the results should look like, not only if the circumstances are what you believe but also if, say, the aerodynamics or the control system have values higher or lower than you expected within reasonable limits; one should know what the effects would be if, say, one of the instruments, the gyroscope or the accelerometer, has an unusually low damping; what the symptoms of a sluggish actuator will be; what effect variations in the supply voltages in the missile will have; and so on. I know that this will sound a tedious and rather prohibitive list of variants. Believe me, it is much better to have some pretty good opinion formed as to these effects before the firing than to try to unravel after the firing what has happened.

I am accustomed to comparing this phase of the work with the use of a map. If you only get the map out when you are lost, you can convince yourself of almost anything, but if you are watching the map as you proceed along the road and say to yourself, "In five minutes I will come to a crossroads with a church to the right and a pond to the left," then you will proceed with much more surety.

I try to get the analysis discussions and conclusions on the firing of a particular round cleared up in a fortnight, perhaps three weeks at the outside. I am afraid that, unless this is done, the whole momentum of the trials is rapidly lost. As I said in the beginning, time is all important in weapon development. There is a tendency to make something of an exact science out of guided weapon development. Maybe it can be made an exact science, although I doubt it. The point is, there just is not time to analyse fully and appreciate all the information coming from any single trial, let alone from a number of trials. One can only hope to make the best decision possible in the time available with the information available. Frankly I do not think this is so difficult. The procedure I follow is that the raw data from the trial, generally in the form of telemetry records, velocity plots and perhaps some rough kinetheodolite figures, are distributed to the half dozen or dozen or so people concerned. The Project Officer holds a short meeting from which emerge the points on the records which really do warrant a more detailed analysis, using the accurately calibrated information, and so on. From a comparison of what the forecast results should have looked like against the actual fired results, it is not too difficult to see where the anomalies are and what needs explaining.

This is clearly a task for the experts in the relevant Design or Trials Branches. I then hold a formal discussion on the round some ten days or a fortnight after the firing. Representatives from all the branches of the Guided Weapons Department are present, and a story has to be produced which will stand up to criticism from anybody round the table. It may seem rather wasteful in time to have an aerodynamicist sitting there while an electronics engineer explains precisely what happened to the guidance system, but, believe me, I have found it most valuable. There is no man so blind as the expert in his own field. An intelligent man, working in some other part of the missile and knowing the system, has often seen the answer where the expert has failed. Some order and proportion can be got into

the proceedings if you remember that no piece of mathematics is any better than the basic data and assumptions from which it started; get all the facts on the table; get back to the first principles behind the phenomena; and perhaps above all, try to check conclusions by a different line of argument.

Do not let me give you the impression that I regard the development of a guided weapon as being in any way easy and straightforward. Indeed, if I did not get difficulties in the development of a weapon, I would be pretty worried about it. A guided weapon development which had no serious difficulty would clearly be such a small step forward that, however great its technical success, it would be of little military value. However, there is a deal of difference between banging your head against a brick wall and climbing over it.

So much for general matters. I would like to spend the remaining time on three or four specific problems to illustrate the type of problem that one runs against.

4. Assessment and Design Study

First of all, I would like to consider a problem typical, I think, of the sort that one meets in the assessment stage. Let us take the case of a radar homing weapon to be fired from the ground against aircraft targets. Somebody will have laid down the maximum height and range wanted from the weapon. If at this stage you care to make one big guess, the rest I think will follow fairly logically, but it is a most important guess. It is the weight of warhead needed to ensure adequate lethality against the assumed threat. Having given, as we have said, the maximum range and height, it is the warhead weight that very much determines the overall weight of the missile. Every pound increase of the warhead weight may mean an increase in missile weight of 3 lb. Now the effectiveness of the warhead depends very much on the miss-distance. Strictly speaking, the quantity that matters from the point of view of warhead lethality is the distance between the missile and the target at the moment of fuse detonation.

But, if we leave out of the argument problems about the correct time to initiate the fuse and so on, we are left with miss-distance as defined as the closest approach between missile and target. Now despite all that one sees on the films and all that one reads about in glossy magazines, I do not suppose it is any great breach of security regulations to say that it is not the usual thing for the missile to hit the target. For a variety of reasons the missile is likely to miss the target by varying amounts, depending on the particular conditions and manoeuvres of the target for the particular engagement.

The miss-distance, in general, will be made up of a bias component and a random component, the two being expressed in statistical language as the "mean" miss and the "root mean square" miss-distance. There are a number of causes of these: we have time to consider only a few of them.

For example, in a beam-riding missile the arrangement may well be that the missile develops an acceleration towards the beam centre proportional to the distance of the missile from the beam axis. To take concrete figures, the arrangement might be that there is an acceleration of $1g$ for every 30 ft. distance of the missile from the beam. Hence, if the beam is having to move at an acceleration of $2g$ in order to hold the target at the time when the missile is in the vicinity of the target, then the missile will be lagging the beam by some 60 ft. and, assuming that the beam is holding the target accurately, there will be a bias miss-distance of 60 ft. A miss-distance also will arise when the missile is physically incapable of developing the accelerations kinematically necessary to intercept the target. This is a situation likely to arise, for example, at the end of a homing course interception. If, in the final phase, the sheer kinematics of the situation demand an acceleration of, say, $15g$ to collide with the target, and the missile is only capable of $10g$, then the missile must miss the target by some distance or other, depending on the circumstances. Similarly, sluggishness in missile response will cause a miss-distance, even if the missile is capable of developing the demanded g, if it cannot develop it fast enough, then it will not give a true collision.

Among the causes of random miss-distance, the principal one is what is commonly called "radar noise." That is to say, in addition to the genuine guidance signal to the missile, whether it be the signal it picks up in the field of a radar beam in the case of a beam-rider, or the reflected signal entering the homing head of a homing missile, there is an unwanted random signal, which we call "noise."

There are a number of causes of this noise, and in addition, of course, to the noise coming from outside the missile there are spurious signals of one sort or another generated in the electronic equipment carried by the missile itself. Vibration of the missile itself, for example, by shaking valves, whether they are electronic valves or hydraulic valves even, can generate spurious noise signals. For simplicity in the argument, I want here to consider the external radar noise.

If we consider our example of a radar homing weapon, the information that we are using to steer the missile comes from reflected energy from the target. There are two sources of randomness in this. First of all, from instant to instant the target as a whole will have a variable coefficient of reflectivity. One moment it may be a comparatively good reflector, the next moment a poor one. This instant-to-instant variation in strength of the returned signal into the homing head can cause a spurious random signal into the homing system. But, in addition to this reflectivity effect, there is also the point that the centre of gravity, as it were, of the area of reflection can vary. We can visualise this simply, that at one moment the echo appears to come from the tip of one wing, the next from the tail, the next from the other wing, then from the body, and so on, varying from instant to instant in a random fashion over the target aircraft structure.

Now the popular way of steering a homing weapon is to use what is called "proportional navigation." That is to say, the missile homing head determines the sightline between the missile and the target, and computes in one way or another the rate at which the sightline is turning in space. The missile is then caused to turn in the same direction as the sightline at a rate proportional

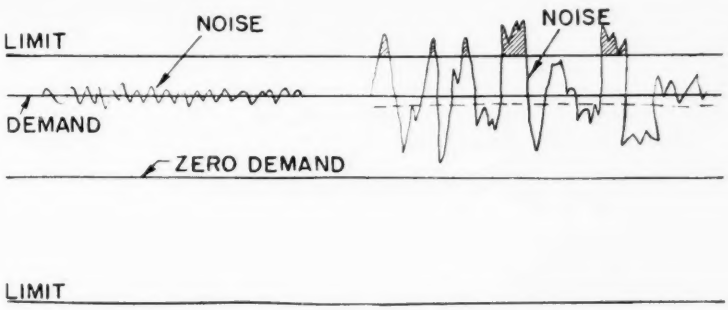


FIGURE 1.

Diagram illustrates a demand signal, say for acceleration going through a system with limits. On the left hand side the superposed noise is small so that the combined signal does not exceed the limit. So nothing is lost and the *mean* value of the signal is unaltered. On the right hand the superposed noise is large enough to cause the combined signal to exceed the top limit. Hence some signal is lost in the upper half and the *mean* value of the signal is lower than expected.

to the sightline rate. That is, $\dot{\theta}=\lambda\dot{\sigma}$ where θ is the angle of the missile flight path, σ is the angle of the sightline, both measured with respect to some datum fixed in space, and λ is the constant of the proportional navigation. It can be shown that, if the missile speed is great enough relative to target speed, then kinematically if the constant λ is greater than two, this law eventually brings the missile on to a collision course against a non-manoeuvring target. Moreover, I think one can readily see that, on the simple kinematical theory, the larger the constant λ, the more rapidly does one wipe out initial launching errors.

Now the difficulty is this, the radar head cannot measure the direction σ of the sightline accurately; on average a good radar head will, over a sufficiently long period of time, have the true mean value for the sightline, but there will be from instant to instant random fluctuations about this, arising in the way that I have described earlier. Now the effect of this random fluctuation of sightline is made worse because the homing system operates on the rate of change of sightline and this can lead to quite a serious noise signal in addition to the genuine required signal.

This noise has four principal defects of varying degrees of gravity. The noise in the first place can lead to spurious acceleration demands on the missile; this means that the missile is moving its control surfaces quite unnecessarily, thus dissipating the power supplies carried on the missile, increasing the missile mean drag and thus decreasing the range or velocity of the missile, and also directly causing the missile to miss the target by some distance.

The fourth effect is a slightly more complicated one. It is what I believe is called "loss of stiffness because of noise in the presence of limits." It is best illustrated by a diagram (Fig. 1). Every missile will have some limit of acceleration which it cannot exceed or cannot be permitted to exceed. If the genuine guidance signal calls for a missile acceleration and there is superimposed on this an unwanted spurious demand for acceleration of a random nature arising from noise, then it may be, and indeed generally is the case towards the end of a homing trajectory, that the total signal of genuine guidance signal plus noise demand exceeds the acceleration limits for appreciable intervals of time. Then the mean value of the total signal of wanted signal plus unwanted noise will be lower than the true wanted signal. This means that the missile will not develop the average acceleration necessary to intercept the target, and there will be a miss of the "bias" nature, in addition of course to a random error of miss contributed by the noise on its own behalf. The amount by which the system is degraded can be very serious if the noise is large, and particularly if there is a steady demand for missile acceleration anywhere near the missile limits.

The noisy signal can be prevented from getting through to the missile control system by the device of inserting a filter, either at the output of the guidance equipment, or usually somewhat farther forward towards the homing head. But any filter must operate not only on the noise but also on the genuine wanted signal. If the noise were all at high frequency and the wanted signal all at low frequency, then the situation would be quite satisfactorily handled by a filter which virtually stopped all the high frequencies getting through but allowed the low frequencies through.

Unfortunately, like so many things in engineering, there is an overlap region where both the wanted signal and the noise have frequencies which interest us. Thus the sort of filtering network that is employed must be a compromise between cutting out the unwanted noise signal and not cutting out too much of the wanted guidance signal.

It can be looked at another way, which is perhaps more profitable to us in the present context. That is to say, any filter is essentially a smoothing device, an averaging device. As such, there is an element of time delay about it. If you like, the filter waits to see whether the signal is a rapidly changing one, in which case it averages the thing out to zero and, therefore, of no account, or it waits to see whether it is a gradually changing one, which it then permits to pass as a genuine demand. However, from our point of view a filter introduced to remove the noise has the unwanted side effect of introducing a delay in a genuine signal. This delay is a delay in the missile developing the acceleration required by the kinematics for collision. Therefore, as I said earlier, it contributes something to the miss-distance.

It is the noise problem also that puts a limit on the size of the proportional navigation constant λ that can be employed in the equation above. Clearly the larger λ, the larger the contribution of noise going into the missile. In practice one may compromise on a λ of approximately four and some filtering of some sort. For a target executing little manoeuvre the value of λ has no importance on miss-distance, except at short range. The bigger λ, the faster the launching error

direction is corrected. So that it is only with targets at the closest engagement range that λ is of really great significance. With targets capable of large manoeuvre the size of λ may be important even at the end of the flight. However, what I have tried to show is that by the physical nature of things there is bound to be, in the general case, a miss-distance and that the designer is forced to make some compromise to arrive at an optimum.

The matter does not end here. We can determine from the foregoing elementary considerations what the miss-distance is likely to be, and from this determine the size of warhead. But we must now go round the course again with considerations like the following. The radar dish of the homing head must have a cover on it, a radome. Clearly, from the aerodynamic point of view, a nicely shaped nose of slenderness ratio something like 3 : 1 would be what we would like aerodynamically. But the radar sightline suffers some aberration on passing through the material of the radome. This amount of aberration will not be constant for all angles of look through the radome, as you can well imagine. It is not difficult to guess that from the radar point of view a hemisphere would be more like the right thing, because it would look very much the same to the radar dish whatever the angle of look. Now the bad effect of this radome aberration is this. The missile flying under control to intercept the target is constantly manoeuvring and changing its direction. In the process of doing this, it will be oscillating more or less in pitch and yaw at the missile weathercock frequency. This is perhaps 1 c./sec. Remembering that the radar sightline, apart from the radar noise, is approximately constant in direction on a collision course, this pitching and yawing of the missile means that the radome is being moved up and down, left and right, across the radar beam. As the radome is wobbled about in this manner, the amount of aberration of the beam will vary according to the particular part of the radome the beam is looking through. That is to say, quite spurious changes of sightline are fed into the homing guidance equipment, arising not through any genuine movement of the target but from movements of the missile body itself. These spurious signals which will be mainly at the frequency of the missile pitching and yawing, say the 1 c./sec. mentioned, can get through the guidance system and be presented to the control system as demands for acceleration. It is not difficult to see that a state of resonance could be set up with a demand of this frequency causing the missile to oscillate more violently at that frequency, thus increasing the amount of spurious signal due to aberration and so on round the cycle, causing the missile to oscillate violently. Even if the effect were not sufficient as to be catastrophic, it could readily be a serious contribution to the general noise signal referred to.

The designer is faced with another compromise that he must make. It arises in this way. Increasing the fineness ratio of the nose will decrease the drag and therefore either increase the range for the same amount of propulsive power, or will decrease the weight of the missile for the same range, because one can reduce the amount of propellant carried. We have assumed that the range has been laid down for us; therefore the effect that we get in increasing the nose fineness ratio will decrease the weight of the weapon from the aerodynamic point of view. However, increasing the fineness ratio increases the amount of aberration variation. This means that more filtering must be put into the missile control system to prevent the parasitic oscillations referred to already. The effect of putting in these filters or lags into the missile control system in order to get stability means, as shown earlier, that there is a lag in responding to genuine demands for manoeuvre. Hence the miss-distance will go up; hence, if we are to maintain the same lethality, the warhead weight must be increased. But if the warhead weight is increased, the all-up weight of the missile must be increased by approximately three times the change in warhead. Thus, overall, if the range of the missile is maintained constant and the lethality is maintained constant, the increase of fineness ratio of the nose may indeed lead, not to a decrease in weight as one would imagine from simple aerodynamic arguments, but to an increase in weight.

On the other hand, decreasing the fineness ratio of the nose so that it tends more to the hemispherical, although this decreases the troubles arising from aberration, that is to say reduces the amount of filtering necessary in the missile and hence reduces bias errors arising from delay in responding to demands, the blunter the nose the more the drag will go up, and hence the more propulsive power required to carry the missile to the same range.

5. Problems of Fast Response and Stability

In the example considered, we have touched upon two things. The first is the need for a fast response from the missile and the second is the need for stability of response. In addition to these, there is also a need for the response of the missile to be largely invariant to external conditions. That is to say, that a demand for a certain acceleration of, say, $4g$ should be met by a response of approximately that amount irrespective of the speed of the missile or the altitude at which it is operating, within wide limits. These three requirements are met not by any sophisticated aerodynamic design of wings or fins, but by what are known in the jargon of servo-mechanism theory as "feedback" and "shaping" networks in the control system.

It is perhaps a sad thing to say before this Society but, in guided weapon design, the aerodynamic matters are not at all critical. Provided that one has wings of suitable area to develop the required acceleration at the top altitude without undue incidence and fins capable of developing a reasonable amount of moment under the same conditions, then the control engineer ought to be able to look after most of the rest. The concept of feedback is an important one.

I would first like to give an example to show its importance, and then to deal with some of the problems that arise simply because one has used feedback to solve some problems. I am afraid, as in all other activities, you get nothing without paying some price for it.

Consider first an example where no feedback is

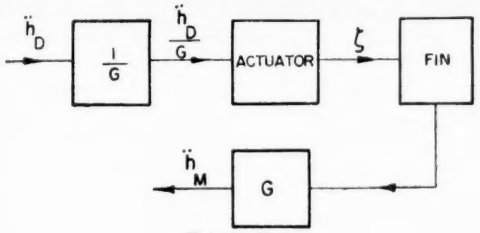


FIGURE 2(*a*).

employed. Suppose we have a missile which, for a given altitude and speed, can develop an acceleration of G per unit deflection of the rudder. I am thinking here of a cruciform missile and just considering one plane (Fig. 2(*a*)). Then a demand from the guidance system for an acceleration $\ddot{h}_D$ could be met in the following way. The electrical signal representing $\ddot{h}_D$ could be fed to a black box which would divide the quantity by G, because we want one unit of rudder for each G units of acceleration. This would then be the demanded fin angle and, as we have assumed that the missile develops G units of acceleration per unit of fin, the resultant missile acceleration $\ddot{h}_M$ would be $G\frac{\ddot{h}_D}{G}$. That is to say $\ddot{h}_D = \ddot{h}_M$, or the actual developed missile acceleration equals the demanded acceleration and all looks to be well. But all will only be well under one set of conditions. That is under the conditions for which the value of acceleration per unit rudder is precisely G. In actual fact, the amount of acceleration per unit of rudder will vary with missile speed and altitude and with various other factors. So that the symbol G appearing in my diagram after the fin is, in fact, a quantity varying with missile speed, altitude, and so on, so that we ought to call it, say, G'. Thus if the missile has been set up for some mean value of this quantity G, the actual missile acceleration is in the ratio to the demanded acceleration of G'/G. Thus the actual acceleration varies with the external conditions.

But if we use feedback, the situation is altogether different (Fig. 2(*b*)). The method is to carry an accelerometer in the missile and use this to measure the missile acceleration $\ddot{h}_M$. This measured acceleration is fed to a differencing element in which the demanded acceleration $\ddot{h}_D$ has subtracted from it the actual acceleration $\ddot{h}_M$ and the difference or error is then fed to an amplifier of gain K, the output of which is used as a demand for fin angle. Thus the fin angle demanded, we can say, is $K(\ddot{h}_D - \ddot{h}_M)$. If, as before, the missile acceleration per unit of rudder is G, then we have $\ddot{h}_M = KG(\ddot{h}_D - \ddot{h}_M)$ which, if you solve for $\ddot{h}_M$ gives

$$\ddot{h}_M = \frac{KG\ddot{h}_D}{1+KG}$$

The larger K is made, the nearer the ratio $\ddot{h}_M : \ddot{h}_D$ approximates to unity. Moreover, with a large K the fraction is relatively insensitive to quite large variations in G. Unfortunately, there are practical limits to the sizes of the gain K that we can employ. They arise

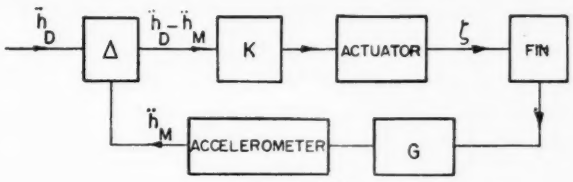


FIGURE 2(*b*).

in this way. The response of the missile to rudder movement, as can be seen from an examination of the aerodynamic equations, is a function not only of the amplitude of the rudder movement but also of its frequency. If you look at this block diagram, we have in it the same sort of loop as arose in talking about the aberration problem.

I think that it is not difficult to see that self-sustained oscillations could be set up round the loop from the fins through the feedback and the differencing element and the amplifier to the fins again. This is particularly so when you consider that in the practical case there will be a lag in the actuator moving the fins and a lag in the accelerometer which measures the acceleration $\ddot{h}_M$. In fact there will be a frequency in which the sum of the lags arising from the actuator, the aerodynamics and the accelerometer add up to precisely 180°. If, in going round the loop at this frequency, the total loop gain is greater than one, then K is too large and some smaller value must be chosen leaving an adequate margin of stability.

You will see then from this elementary example that, to get the desirable high amplification K in the loop so as to make the missile control less dependent on variable outside quantities, it is essential to reduce in some way the lag in the various elements of the loop. The elements in the loop are those already discussed, namely the missile control surface actuators and associated servomechanisms, the missile aerodynamics, and various instruments such as accelerometers and gyroscopes.

6. Improving the Aerodynamic Response

Obtaining actuators and instruments of sufficiently high response is a relatively straightforward design problem. I want to look here at the aerodynamic side to show how the problem of getting faster response out of the missile's natural aerodynamics is achieved.

Consider motion in the yaw plane only. In order to develop a transverse acceleration in this plane, the rudder is in the simplest case first deflected through the angle required to give the final steady state transverse acceleration. This sets the rudder at an incidence to the air flow, thus developing a force on the rudder which gives a moment about the c.g. of the missile, thus causing the missile to rotate about its yaw axis. This moment causes an angular acceleration about the yaw axis, depending of course in magnitude on the magnitude of the moment and the size of the moment of inertia of the missile in yaw. However, the point is an angular velocity in yaw is set up so that the missile is rotated initially with increasing velocity about its yaw axis.

This rotation of the missile body in yaw sets up an

angle of incidence between the missile wings and the direction of motion of the c.g. As a consequence of this, a transverse force is built up on the missile wing; there is also, of course, a component from the aerodynamic forces on the missile body, but for simplicity we will ignore them, or more properly lump them in with the wing effect.

In a stable missile the arrangement is such that the force set up at the wings is slightly forward of the missile c.g., so that as the incidence is generated a moment is generated about the yaw axis opposing the yawing moment set up by the rudders. Eventually the state is reached, called the "trimmed condition," when the yawing moment on the rudders is just balanced by the opposing yawing moment on the wings. In the case of rear controls, as the missile builds up incidence in yaw, the incidence on the rudders is decreased so that the moment at the rudder decreases from its initial value down to some value in the trimmed condition. You will see from this sequence of events that there is a lag between applying the rudder angle and the developing of the steady state missile acceleration. Generally speaking, in missile practice this lag is too great to be acceptable. Worse than that, however, the missile does not in its natural condition reach the steady, trimmed condition in one go, as it were, in the smooth manner I have rather suggested.

The point is that under the moment of the rudders the missile builds up a yaw angular rate; this is the quantity usually denoted by r, so that when the yawing moment falls to zero, when the moment provided by the wings equals that provided by the rudders, the missile yaw rate r is actually at a maximum and the missile overswings its trimmed position. It swings over until the negative acceleration developed by the wing moment overpowering the rudder moment eventually brings the missile to rest in yaw and swings it back again. We then, in this simple picture, arrive at a situation where the missile oscillates indefinitely about its trimmed position. This is what is called the "natural weathercock frequency" of the missile.

In order to damp this motion down so that the oscillations eventually decay away and the missile settles on its trimmed position, it is necessary to introduce a moment about the yaw axis which is proportional to rate of yaw and is opposing it. There is a certain amount of natural damping of this nature present because, as the missile rotates about its yaw axis, the wings and fins are rotated through the air stream and encounter some resistance in doing this.

This resistance provides, as I think can be seen readily, a moment which is, to a first approximation anyhow, proportional to the rate at which the missile is yawing. In the case of an aircraft the aerodynamic surfaces are sufficiently large for the damping forces set up in this way to be, in general, sufficient to provide natural damping for the aircraft. This is not the case, in general, for a guided weapon. To keep the drag of the missile as low as possible and to avoid complication in storage and in launching the missile, we generally aim to keep the surface areas as small as possible compatible with developing sufficient acceleration under the worst conditions. It turns out then that the amount of natural damping present in a guided missile is quite inadequate to give a sufficiently damped motion. We must therefore find some other way of doing this.

There are two things, therefore, that we wish to do to the missile aerodynamics; the first is to provide some synthetic damping, the second is to take some steps to improve the speed of response between rudder movement and the development of the steady state acceleration. To achieve the first we must set up a moment opposing rate of yaw and proportional to the rate of yaw. This we do by the use of the rudders, by giving them a deflection proportional to yaw rate at any instant and in opposite sense. This can be done by carrying in the missile a rate gyroscope so situated as to record the yaw rate r of the missile. A signal can be taken from this, multiplied by a suitable scaling factor and sent to the servo-mechanism controlling the actuators so as to provide a component of rudder movement opposing yaw rate. We can increase the speed of response of the missile by causing the rudder initially to deflect through a larger angle than is finally required.

For example, if the rudder is moved initially through five times the final steady state angle of rudder required, then the moment in yaw initially developed will be five times what it would otherwise have been. We avoid ending up at five times the required transverse acceleration by the device of taking off the rudder as the required acceleration is built up. That is to say, rather on the lines of the example I gave earlier, a signal proportional to missile acceleration is fed to the actuators in such a sense as to back off the amount of fin applied. This acceleration can be recorded by an accelerometer which ideally must be placed at the c.g. of the missile, otherwise it will include terms involving angular acceleration which might be tiresome.

I have worked out some figures for a quite hypothetical missile flying at a speed $M=2$ at 50,000 ft. In the unmodified state the natural weathercock frequency is 0·65 c./sec. with very little damping, and the steady state acceleration per degree of rudder is $1{\cdot}25g$/degree. If a rate feedback of 0·2 rads. of rudder/radian/sec. of yaw rate is introduced, and an acceleration feedback of 0·0005 rads./ft./sec.^{2} of transverse acceleration is also applied, then the natural frequency in the weathercock mode is increased to 1·15 c./sec. (that is, the response rate is almost doubled) and the damping is increased considerably to a damping factor of 0·76. (This means that the damping is rather less than critically damped, and is about the sort of response we require.) The price we pay, of course, is that a smaller portion of the rudder deflection is now available for developing the steady state transverse acceleration, the figure for which is now $0{\cdot}53g$ per degree.

There are again limits to the amount of feedback that can be employed before running into instability troubles of one type or another. You can see, in the same way as we discussed before, that there is always a risk of an oscillation being sustained, either round the rate gyroscope loop, or round the accelerometer loop. Indeed, one of the important problems in the design of a control system is the correct characteristics

and siting of these instruments in the missile. For example, application of rudder imposes a force on the missile at the rudder position. This will set up the bending oscillations of the missile as a beam. These flexural oscillations will, of course, be recorded by the accelerometer and rate gyroscope, just as the genuine rigid body motions of the missile will be recorded. If the accelerometer is placed at an anti-node of the flexural oscillation, then the effect will be at its worst; if placed at a node; then the accelerometer should register nothing at all from the flexural motion. Similarly, a rate gyroscope at a node will be worst situated, because at a node the change of direction of flexural motion is fastest. If the instruments can be sited at node and anti-node respectively, then there is no problem. This is not always the case because of the construction of the missile. An accelerometer situated in the middle of the rocket motor would not be entirely happy.

Care must therefore be taken in siting the instruments so that oscillations at the flexural frequency, or in any way associated with the flexural frequency, cannot be sustained round either the accelerometer loop or the rate gyroscope loop. If the instruments cannot be sited properly, then the introduction of some filter or shaping network in the loops should shift the phase of the oscillation sufficiently to prevent instability.

7. Cross-Coupling Effects

I have passed over a whole number of control problems; I feel I ought just to mention them as, in fact, I have found that they form the bulk of day-to-day problems. These are problems associated with backlash, stiction, vibration, missile aerial polar diagram, flame attenuation, and the multitudes of cross-couplings that can arise between the roll, pitch and yaw planes. These last form such an important group and seem so little understood that I would like to spend a minute on them.

If you turn to any of the classical textbooks, such as Lamb's *Higher Mechanics*, you will find the aerodynamic equations for aircraft (and this includes missiles) flight written in full. (*See* Appendix.) It is most important when one has been round the design of the system once, or certainly after one has been round the system twice, to go back and look at the fundamental equations. For this reason, we have been so accustomed over the years to rejecting certain terms involving the products of the velocities and rates (because they made the mathematics unmanageable) that we are in danger of forgetting about them altogether. These terms can in some circumstances be most powerful.

For example, it is customary to roll-stabilise a missile, or at least to hold the roll rate, at some low figure of $\frac{1}{2}$ c./sec. say. This means that all the terms involving roll rate in the equations for transverse motion can be neglected. This is not true during the actual phase of roll stabilisation itself. Then the roll rate can be high, also the sideslip can be considerable. The de-stabilising effects of this cross-coupling term can, moreover, be aggravated by the fact that the fins just cannot move fast enough to cope with roll stabilisation and with ensuring stability in the lateral planes. It is important too to look at the aerodynamic derivatives and see that we really have been justified in throwing away as many as we do. For example, at high incidence, when the missile is manoeuvring in both pitch and yaw, it is extremely likely that there will be fairly significant rolling moments of the type L_v. A glance at the full fundamental equations shows that this might well be serious and lead to an instability between roll, pitch and yaw. There is really little excuse though for our being caught out by this type of problem. They arise directly from Newton's Laws and, if we walk into the trap, we have only ourselves to blame for not having studied the problem properly.

I now want to go back to my first consideration, that is the miss-distance, and show how starting from this the parts of the system depend on each other. Take, for example, a beam-rider. We decide, say, that if the missile develops an acceleration of 1 ft./sec.^{2} for every foot of distance from the beam, then the miss-distance will be acceptable. This means that the control frequency of the missile in the beam, commonly called the "weave" frequency for a beam-rider, is 1 rad./sec. We assume that it has been proved that this is fast enough to secure a sufficiently small miss-distance, but is too slow to give much trouble from noise, and so on. The next step is to introduce a damping term proportional to rate of change of error from the beam, to prevent the missile hunting indefinitely about the beam axis. In this case if we add the demanded acceleration, a term $1{\cdot}4\dot{h}$, we shall get a weave motion damped by a factor of 0·7. This demanded acceleration of $h+1{\cdot}4\dot{h}$ can be got by phase advancing the output of the guidance receiver.

Having determined what these outer loop parameters, as they are called, should be to obtain a satisfactory miss-distance, we must now determine the characteristics necessary in the inner missile control loop to match the required performance of the outer loop. For a roll-stabilised beam-riding missile, we can write down a number of frequencies in ascending order, starting with the weave frequency. They are: —

(*i*) Weave frequency,
(*ii*) Missile modified weathercock frequency,
(*iii*) Roll stabilisation frequency,
(*iv*) Actuator frequency,
(*v*) Instrument frequencies.

There are also, of course, other frequencies such as those inside the guidance equipment; for example, the frequency of the automatic gain control, and for a homing missile a range of frequencies associated with the servo systems of the homing head. Keeping to the present example of a sequence of control frequencies, what we require is that the next frequency up the scale should be sufficiently high relative to the preceding one so as not to degrade in any marked extent the performance of the previous part of the system. For example, the overall beam-riding response, discussed before and expressed in the equation $\ddot{h}_M = h+1{\cdot}4\dot{h}$, has assumed a missile of infinitely fast response, so that the missile acceleration is developed as soon as the demand signal (the right-hand side of the equation) is presented to the

control system from the guidance receiver. This we know will not be the case. There is bound to be a delay of some sort between the demand for acceleration and actual achievement of acceleration. What we aim to do in the practical case is to see that the missile modified weathercock frequency is sufficiently high that the kinematic equation is not seriously affected.

Now to a first approximation the frequency responses of the missile, the actuators, and the instruments are those of the quadratic transfer function. That is to say, the response is given by the reciprocal of

$$\left(1+2c\frac{p}{\omega_N}+\frac{p^2}{\omega_N{}^2}\right)$$

At the resonant frequency ω_N the phase shift of this is obviously 90°. For a frequency one-fifth of the resonant frequency the phase lag for a reasonable damping factor of 0·7 is about 16° and the amplitude is only slightly decreased by about two per cent. That is to say, a missile whose modified weathercock frequency is five times the wanted weave frequency will have little effect on the amplitude of the weave frequency and will only shift the phase of that frequency by 16°. If the ratio of the frequencies is as low as three, then the phase shift would be nearly 30°.

This suggests that, if the natural frequencies ascend by a ratio of five between adjacent frequencies, then there should be satisfactory isolation of one part of the system from the next. A larger factor for the ratio would give an improvement but this raises other problems of sensitivity to noise and so on, because the frequencies at the top end of the scale would be getting high indeed. On the other hand, it would probably be not at all advisable to reduce the ratio below three.

Using this ratio of five, for purposes of example, we have for the case where the basic weave frequency was 1 rad./sec. the following scale of frequencies: —

(*i*) Missile weave frequency in the beam
5 rad./sec. (approx. 1 c./sec.)

(*ii*) Roll frequency
25 rad./sec. (approx. 5 c./sec.)

(*iii*) Actuator frequency
125 rad./sec. (approx. 20 c./sec.)

(*iv*) Instrument frequency
625 rad./sec. (approx. 100 c./sec.)

These higher frequencies may not be easy to obtain. Moreover, they are likely to bracket the fundamental flexural frequency of the missile structure and other important flexural frequencies.

Having settled the basic frequencies of the system, we set about checking whether the system will still be satisfactory when the second order effects, ignored up to the present, have been allowed for. The first thing to check is that the system is still satisfactory under conditions (these are generally at the beginning of flight at boost separation or towards the end of flight at high altitude with large accelerations in both planes) when the cross-coupling terms in the basic kinematic equation become important. This will generally lead to some changes to the system, such as increasing the size of the wings, or introducing some shaping networks to alter the frequency of some part or other of the auto-pilot system, the introduction of limiting devices of some sort to prevent saturation in some part of the system, and so on.

Next one examines all the possible sources of unwanted feedback of one sort or another. An obvious one to examine, and one so obvious that it probably is considered in the very design of the control system itself, is the effect of the missile flexure on the control instruments and the performance of the inner loops of the missile. At the beginning of this paper I dealt with another feedback loop of an unwanted nature, that was in connection with radome aberration in a homing missile. In the case of a beam-rider the ionised gases from the rocket motor may cause attenuation of the guidance signal received by the missile from the beam. If this were just a constant attentuation it would not matter significantly. However, as the missile moves about in the beam, pitching and yawing, it may be that the angle of look through the attenuating gas is varying so that the incoming signal to the missile is modulated by the missile's own movement. It is true that the automatic gain control of the missile guidance receiver is there to look after variations of signal strength, but even so the situation needs examining to see whether undesirable oscillations can be sustained round the loop of missile attitude, rocket motor efflux, attentuation of signal, spurious demand for control fin movement, and back again to missile motion.

8. Conclusion

For all that I have said, I want to come back to the point that I made at the beginning, that is that the real difficulties of guided weapons I have found to lie in the detailed engineering of the various parts that make up a system. Remember, that in dealing with the missile I have only dealt with one component of a guided weapon system. The design and integration of the associated radar equipment, computors, launching and handling equipment, and test gear, presents quite a challenge when the work has to be carried out against a time scale such as is required for a Service weapon.

APPENDIX

CONTROL EQUATIONS

NOTATION

mass	m		
missile axes	roll	yaw	pitch
forces	X	Y	Z
moments of inertia	A	B	C
moments	L	M	N
velocity of c.g.	U	v	w
rates about axes	p	q	r
control surface angles	ξ	η	ζ

FUNDAMENTAL EQUATIONS

$$\dot{U}+qw-rv=\frac{X}{m} \quad . \quad . \quad . \quad (1)$$

$$\dot{v}+rU-pw=\frac{Y}{m} \quad . \quad . \quad . \quad (2)$$

$$\dot{w} + pv - qU = \frac{Z}{m} \quad . \quad . \quad . \quad (3)$$

$$A\dot{p} + (C - B)\,qr = L \quad . \quad . \quad . \quad (4)$$

$$B\dot{q} + (B - C)\,rp = M \quad . \quad . \quad . \quad (5)$$

$$C\dot{r} + (B - A)\,pq = N \quad . \quad . \quad . \quad (6)$$

FIRST APPROXIMATIONS

The usual assumptions are:—

(*a*) Missile roll-stabilised or at least roll rate low so that terms in p neglected.

(*b*) The forward velocity U remains constant.

(*c*) Derivatives ignored other than:—

$$Y_v = \frac{\partial Y}{\partial v}, \quad Y_r, \quad Y_\zeta, \quad N_v, \quad N_r, \quad N_\zeta,$$

$$Z_w, \quad Z_q, \quad Z_\eta, \quad M_w, \quad M_q, \quad M_\eta,$$

$$L_\xi, \quad L_p.$$

(*d*) Derivatives assumed constant.

Then the fundamental equations give:—

Equation (1): Ignored.

Equation (2): $\dot{v} + rU = y_v v + y_r r + y_\zeta \zeta \quad . \quad . \quad . \quad (7)$

(where $y_v = \frac{Y_v}{m}$, and so on).

Equation (3): $\dot{w} - qU = z_w w + z_q q + z_\eta \eta \quad . \quad . \quad . \quad (8)$

Equation (4): $\dot{p} = l_\xi \xi + l_p p$ (where $l_\xi = \frac{Y_\xi}{A}$, and so on).

. . . . . (9)

Equation (5): $\dot{q} = m_w w + m_q q + m_\eta \eta \quad . \quad . \quad . \quad (10)$

Equation (6): $\dot{r} = n_v v + n_r r + n_\zeta \zeta \quad . \quad . \quad . \quad (11)$

This decouples the motions of roll (given by Equation (9)), yaw (given by (7) and (11)), and pitch (given by (8) and (10)).

SIMPLIFIED YAW MOTION

Typical values for a cruciform weapon with rear controls at height 50,000 ft. and speed $M = Z$ are:—

$$y_v \simeq -0{\cdot}5, \quad y_\zeta \simeq 150, \quad y_r \simeq 0,$$
$$n_v \simeq 0{\cdot}01, \quad n_\zeta \simeq -50, \quad n_r \simeq -0{\cdot}5.$$

Denoting the Laplace operator by s (this amounts to replacing $\dot{v}$ and $\dot{r}$ by sv and sr) and the sideways acceleration $\dot{v} + rU$ by $\ddot{h}$, then (7) and (11) solved for values above give:—

$$\frac{\ddot{h}}{150s^2 + 75s - 47000} = \frac{r}{-50s - 23{\cdot}5} = \frac{\zeta}{s^2 + s + 20{\cdot}25}$$

These give the yawing rate r and the acceleration resulting from a rudder angle ζ.

If the rudder angle is proportional not only to a demanded angle but has a damping term $0{\cdot}2r$ opposing rate of yaw and a response quickening term $0{\cdot}0005\ddot{h}$, then

$$\zeta = \zeta_D + 0{\cdot}2r + 0{\cdot}005\ddot{h}$$

and we have a fourth ratio equal to the others:—

$$\frac{\zeta_D}{0{\cdot}925s^2 + 10{\cdot}96s + 48{\cdot}45}$$

Comparison of the ratios in ζ and ζ_D shows that the speed and damping have been considerably improved.

CROSS-COUPLING

The equations (1) to (6) are, in fact, coupled together not only by the terms in p, but also because of derivatives like L_v. This roll moment due to sideslip will be serious when there is combined pitch and yaw.

SUMMARY OF THE DISCUSSION

by H. C. PRITCHARD, F.R.Ae.S.

DISCUSSION ranged over organisational and technical topics. The need for specialists to appreciate the overall problems involved in the development of a missile was stressed, and Mr. Clemow instanced his design data system as one means of ensuring that one specialist really knew what another was doing. There was a lively exchange of views on the correct approach to the reliability aspect, with Mr. Clemow agreeing that there was room for some fundamental work but insisting that a "reliability empire" was not justifiable and that reliability would depend primarily on the designers, workmen, inspectors, and so on, *i.e.* on first class but normal engineering organisation.

The difficulty of deciding when to fire rounds and the rate of firing was raised, and Mr. Clemow said that no precise guidance could be given. He expressed the view that a half-hearted approach to firing trials did not pay—in general, two identical rounds should be prepared with, say, four others which provided variants of the calculated settings also available, so that a programme which was adjustable on the results of the initial firings could be followed with the minimum delays.

The merits of fixed or moving wing configurations for a radio homing missile were discussed, with the lecturer stating that the moving wing gave little benefit from the aerodynamic and response aspects, was undesirable structurally, provided an important advantage in the case of ram-jet propulsion, and assisted in overcoming the radome aberration difficulty. His advice was to stick to the fixed wings except when some special feature dictated otherwise.

Mr. R. P. Dunning, Chief Superintendent R.P.D., R.A.E., Westcott, remarked that the lecture had not highlighted any problems in his field, and was pleasantly surprised to learn that, with the trend to solid propellants, propulsion was generally reliable and causing far fewer ancillary difficulties. One speaker referred to the effects of access doors on structural rigidity, and provided an interesting suggestion to cure the problem he posed. This gave Mr. Clemow an opportunity to remark on the importance of rigid mounting of accelerometers, gyros, and so on, and of arranging for the natural frequencies of the different items to be well spaced from each other.

TECHNICAL NOTES

A Method for Providing Warning of the Onset of Buffeting, Stalling and Other Undesirable Effects of Flow Separation

by

D. W HOLDER, D.Sc. and H. H. PEARCEY, B.Sc.

(Aerodynamics Division, National Physical Laboratory)

A METHOD is described for providing warning of the onset of undesirable effects produced by flow separation on an aircraft wing. It is based on static pressure measurements near the trailing edge and appears to have advantages over alternative methods.

INTRODUCTION

Recent work has suggested that certain effects of flow separation such as buffeting, aileron buzz and undesirable changes in loading, occur when a " bubble " of separated flow originating at the leading edge of a wing, or at a shock wave on its surface, first becomes sufficiently large to affect the flow at the trailing edge. When this stage is reached, the static pressure p_{TE} begins to fall more rapidly with increase of free-stream Mach number of lift coefficient for the reasons discussed in Refs. 1 and 2. This is illustrated in Figs. 1(*a*) and (*b*) where the quantity

$$\frac{H_0 - p_{TE}}{H_0 - p_0}$$

is plotted against free-stream Mach number and lift coefficient for a 9½ per cent thick symmetrical aerofoil in two-dimensional flow. In this diagram H_0 denotes the free-stream total head, and p_0 the free-stream static pressure.

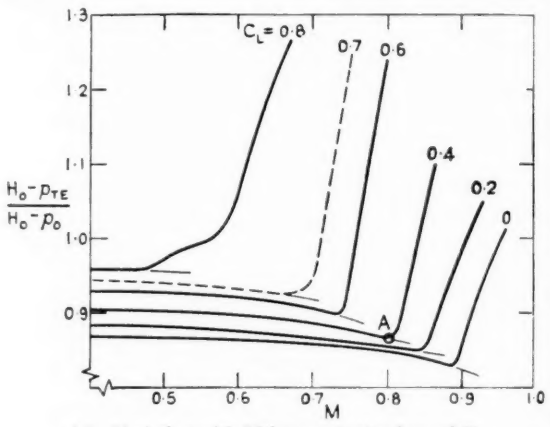
The rapid fall (" divergence ") of the trailing-edge pressure is thus a useful guide to the conditions when separation begins to affect the flow at the trailing edge, and hence (*see* Ref. 3) round the aerofoil as a whole. In particular, it has been found that the locus of the points (plotted on a lift coefficient, or angle of incidence, Mach number basis; *see* Fig. 2) for the divergence of trailing-edge pressure is in good agreement with the boundaries for the onset of certain adverse effects of separation. This is illustrated by Fig. 3 (reproduced from Ref. 4) where the flight-determined buffet-threshold boundary of a delta-winged aircraft is compared with the boundary for the divergence of trailing-edge pressure measured on a wind-tunnel model.

DISCUSSION OF THE PROPOSED METHOD

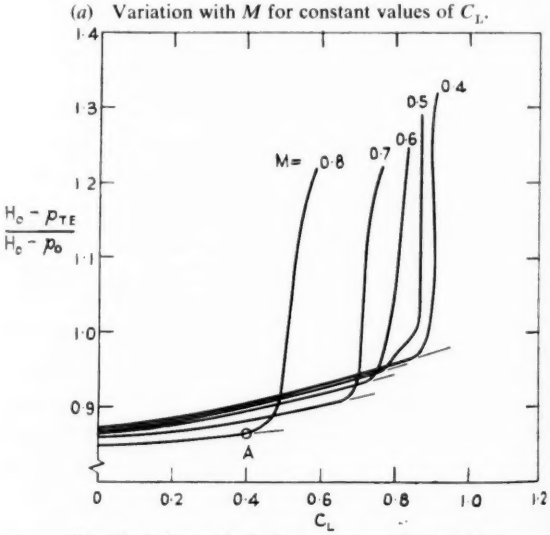
Evidence of the type discussed above suggests that measurements of the pressure at the trailing edge might be used to provide warning when the conditions which would lead to buffeting or other undesirable phenomena are being approached in flight. The details of the mechanical or electrical system that is used for this purpose would need to be decided by instrument designers, but the arrangement sketched in Fig. 4 is given as an example. The pressures H_0 and p_0, and H_0 and p_{TE}, are connected to opposed bellows, each pair of opposed bellows being connected to one end of a balance beam. If the bellows are of equal area, and the opposed pairs are at distances a and b from the fulcrum as shown in Fig. 4, the beam is in balance when

$$\frac{H_0 - p_{TE}}{H_0 - p_0} = \frac{a}{b}.$$

The beam will swing away from the contact shown under the constraint of the spring rate of the bellows when the ratio is less than a/b, and will swing towards the contact



(*a*) Variation with M for constant values of C_L.



(*b*) Variation with C_L for constant values of M.

FIGURE 1. Variation of trailing-edge pressure for typical 9½ per cent thick aerofoil (basic section of the aircraft referred to in *Fig.* 3). (*See* Fig. 2 for locus of " divergence " points " A.")

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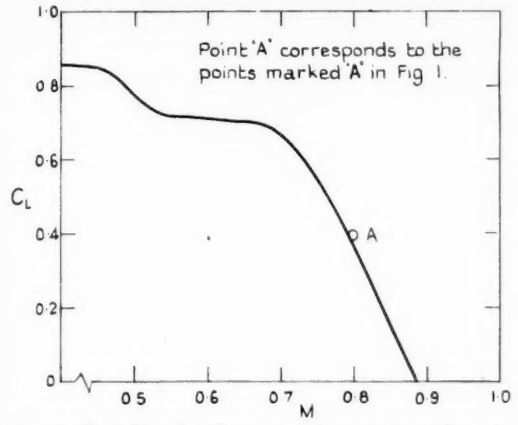

FIGURE 2. Boundary for the onset of separation effects for a 9½ per cent thick aerofoil (locus of divergence points in Fig. 1).

when the ratio exceeds a/b. Thus if the beam is set so that a/b is equal to the value of

$$\frac{H_0 - p_{TE}}{H_0 - p_0}$$

for which the severity of buffeting becomes unacceptable, the contact may be arranged to close and operate the warning system when the ratio exceeds this value. Since the bellows are of equal area, the instrument will be insensitive to changes of ambient pressure. Also, since for a given free-stream Mach number and lift coefficient both H_0 and p_{TE} are proportional to p_0, the system is independent of changes of altitude. The appropriate setting of the ratio a/b would need to be determined by a preliminary flight experiment and, since (*see* Fig. 1(*a*)) the values of

$$\frac{H_0 - p_{TE}}{H_0 - p_0}$$

increase on either side of the divergence point, would have to be sufficiently high to avoid spurious warning signals at low Mach number.

One alternative to this method is to arrange for the warning system to operate when the rate of change of

$$\frac{H_0 - p_{TE}}{H_0 - p_0}$$

or some other function of trailing-edge pressure, with change of Mach number first exceeds a specified value.*

Since the severity of separation is seldom uniform across the span, buffeting usually occurs first when the flow at the trailing edge of a particular spanwise region first begins to be affected, the flow elsewhere on the trailing edge being unaffected. A second method is, therefore, to operate the warning system by an instrument which measures the ratio of the pressures at the trailing edge at two spanwise stations, one being in the region first affected by separation, and the other in a region which is unaffected until higher Mach numbers or lift coefficients are reached.

Whichever system is used, it would be necessary to determine from preliminary wind-tunnel or flight tests the spanwise position of the region where flow separation first

*If this method were used, and the buffeting resulted from increasing lift coefficient, it would be necessary to assume that a change of lift coefficient is accompanied by some change of Mach number.

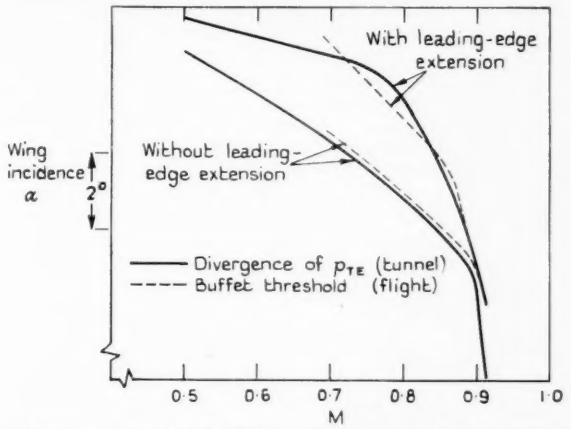

FIGURE 3. Boundaries for the onset of separation effects on a delta-winged aircraft; comparison between wind tunnel and flight tests.

caused buffeting. Alternatively, if one of the first two methods were adopted, it might be possible to use several trailing-edge pressure holes which are scanned by a single instrument, or used to operate several instruments so that warning was given when the pressure first began to fall rapidly at any point along the trailing edge.

If it was found that appreciable buffeting occurred immediately after the trailing-edge pressure began to fall, and earlier warning (in terms of Mach number or lift coefficient) was required, it should be possible to provide this by measuring the pressure on the upper surface slightly ahead of the trailing edge instead of at the trailing edge itself.

BEHAVIOUR WHEN THE FLOW AT THE TRAILING EDGE BECOMES SUPERSONIC

Certain difficulties in the use of the method would arise when the free-stream Mach number is raised to a value that is sufficiently high for the flow at the trailing edge to be supersonic, and these may be illustrated by considering the flow past a two-dimensional aerofoil. Referring to Fig. 5 which includes sketches of the flow past an aerofoil at a fixed positive angle of incidence, suppose that the flow pattern is as in Fig. 5(*i*)(*a*) when the warning device operates. Then with further increase of Mach number the shock wave moves rearwards until it eventually reaches the trailing edge, and becomes inclined to the stream as shown in Fig. 5(*i*)(*b*). Under this condition the tendency for flow separation to occur at the inclined shock is much less than at lower Mach numbers when the shock is nearly normal to the flow. Moreover, even if separation should occur ahead of the shock, the effect on the flow past the aerofoil will usually be less pronounced than at lower Mach numbers. The buffet boundary is, therefore, of the type

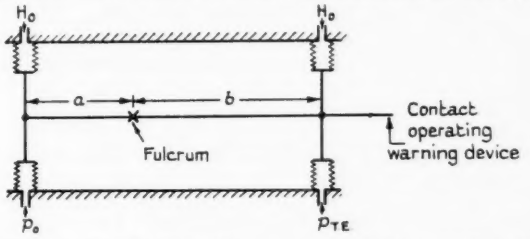

FIGURE 4. Sketch illustrating one method for operating the warning device.

sketched in Fig. 5(*iii*), and if buffeting has occurred at a high subsonic speed, it will frequently disappear at some transonic or supersonic speed. The trailing-edge pressure on the other hand will have fallen to a low value (*see* Fig. 5(*ii*)) when the shock has moved back to the trailing edge, and will remain at a low value with further increase of Mach number. Thus, if the warning system is based on the value of the trailing-edge pressure, it will continue to operate when the shock has moved back to the trailing edge, although buffeting may no longer occur. Since the rate of change of trailing-edge pressure with Mach number is of opposite sign at the two extremities of the rapidly falling part of the curve (*see* Fig. 5(*ii*)), this difficulty would not arise if the warning system were based on the rate of change.

A more severe difficulty arises if the section is designed so that separation and buffeting are absent at all Mach numbers for a certain range of incidence, the buffet boundary being of the type shown by the broken line in Fig. 5(*iii*). Within this range the trailing-edge pressure will still fall rapidly when the shock moves back to the trailing edge, so that a spurious warning will result. Since the values of the angle of incidence for which separation is absent will in general be lower than those for which separation occurs, the fall of trailing-edge pressure which leads to the spurious warning will take place at a higher Mach number than the fall of pressure resulting from separation. A cut-out on the warning system which operated when the Mach number exceeded a certain value might, therefore, provide an acceptable remedy to this difficulty.

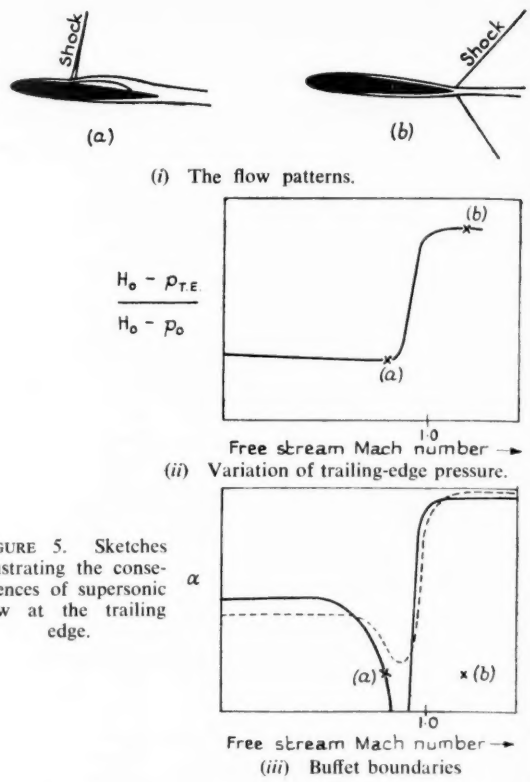

FIGURE 5. Sketches illustrating the consequences of supersonic flow at the trailing edge.

CONCLUDING REMARKS

It is clear that further work would be required before the utility of the suggested method could be reliably assessed. This would involve consideration of the problems of instrument design, and studies of the flow round models of specific aircraft. Whether or not this work would be justified depends on the need which exists for a method which would give warning of the effects discussed, and on the merits of alternative methods.

As far as the authors are aware, existing methods are based on measurements of the angle of pitch of the aircraft, on measurements of the pressure difference between fixed points on the upper and lower surface of the wing near the leading edge, or on the detection of regions of separated flow by mechanical instruments. If the first method is used, difficulties may arise from changes in the calibration of the angle-of-pitch meter with Mach number, and in locating it in such a position that it is not affected by the flow past the aircraft. The latter difficulty seems likely to be particularly severe if it is attempted to measure the local wing incidence in order to provide warning in manoeuvres such as rolling. The second method would overcome these difficulties, but its calibration would also change with Mach number. In particular, the method might become insensitive to the conditions over the rear of the wing at a certain value of the free-stream Mach number for the same reasons that the conditions near the leading edge of a two-dimensional aerofoil become insensitive to changes of Mach number when the Mach number is close to unity.

The method based on the detection of separated flow by mechanical instruments appears to be attractive but, if the instrument is a flap attached to the wing surface and spring loaded so that it lifts when the flow over it is separated, it may be difficult to allow for the effects of changing altitude.

A method based on the pressures at or near the trailing edge should overcome these possible difficulties and, moreover, would be expected to function for a wide range of flight conditions. This is because the divergence of trailing-edge pressure occurs at the stage at which the separation has developed just sufficiently to affect the flow past the wing as a whole (as distinct from a local effect on one surface) irrespectively of whether the separation is associated with a low-speed stall or is induced by a shock wave at some point between the leading and trailing edges.

ACKNOWLEDGMENT

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The Lowest Natural Frequency of an Axial Compressor Blade

by

J. DUNHAM, B.A., G.I.Mech.E., Grad.R.Ae.S.

(Research Department, Rolls-Royce Ltd.)

PUBLISHED WORKS discuss the calculation of the natural frequencies of uniform beam systems, but not simple calculations on beams of arbitrary cross section. This note describes a simple method evolved for estimating the lowest natural frequency of an axial compressor blade, and compares its results with measurements and with other calculations. The logical extension of the method to calculating the frequency of any mode of any cantilever of arbitrary cross section is made in the Appendix.

NOTATION

A	cross-sectional area of blade
E	Young's modulus
g	gravitational acceleration
I	second moment of area about major principal axis
k	constant
l	cantilevered length of blade
R	pitch radius of blade root
x	distance of section from blade root
y	deflection in the plane of minimum inertia
ρ	density
ψ	inclination of the major principal axis of the blade near the root to the compressor shaft. In practice, this is equal to the root stagger angle
ω_0	static lowest natural frequency (rad./sec.)
ω_e	lowest natural frequency of a rotor blade, at speed Ω (rad./sec.)
Ω	compressor rotational speed (rad./sec.)

APPENDIX NOTATION

$y, z, \xi, \eta, \alpha, \theta, c$	defined for a section x in Fig. 3
a, b, e	constants
EI_y, EI_z	flexural rigidities, sometimes written EI_{min}, EI_{max}
GK	torsional rigidity
I_p	polar second moment of area
M_F, M_G	torque loading on section x, about F and G
W_y, W_z	shear loading on section x, in the y and z directions
Y	$c\cos\alpha$
Z	$-c\sin\alpha$

ANALYSIS: STATOR BLADE

In the lowest natural frequency mode the twisting and edgewise movement of the blade are small enough to neglect, so the calculation is confined to deflections in the plane of minimum inertia. When using Rayleigh's method for a uniform beam it is usual to assume an algebraic deflection curve. The essence of the method described in this note is to assume, instead, an algebraic bending moment curve. The boundary conditions to be satisfied are zero slope and deflection at $x=0$, and zero bending moment and shear force at $x=1$. The deflection curve

$$y=k\left(1-\cos\frac{\pi x}{2l}\right) \quad . \quad . \quad . \quad (1)$$

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satisfies the first three boundary conditions, but the simplest curve to satisfy all four is

$$y=k\left[6\left(\frac{x}{l}\right)^2-4\left(\frac{x}{l}\right)^3+\left(\frac{x}{l}\right)^4\right] \quad . \quad (2)$$

The assumption that

$$EI\frac{d^2y}{dx^2}=k(l-x)^2 \quad . \quad . \quad . \quad (3)$$

is more logical for a non-uniform beam, however, and satisfies the third and fourth boundary conditions. For a uniform beam it is equivalent to (2).

Rayleigh's method equates the maximum kinetic energy to the maximum strain energy:

$$\int_0^l \tfrac{1}{2}EI\left(\frac{d^2y}{dx^2}\right)^2 dx=\omega_0^2\int_0^l\frac{A\rho y^2}{2g}dx \quad . \quad (4)$$

But $$\int_0^l EI\left(\frac{d^2y}{dx^2}\right)^2 dx\equiv\int_0^l\frac{d^2}{dx^2}\left(EI\frac{d^2y}{dx^2}\right)y\,dx$$

if $y=\dfrac{dy}{dx}=0$ at $x=0$, so substituting (3) in (4) gives the simple expression

$$\omega_0^2=\frac{\displaystyle\int_0^l y\,dx}{\displaystyle\int_0^l\frac{A\rho y^2}{2g}dx} \quad . \quad . \quad . \quad (5)$$

The values of I and A along the blade can be found from standard tables of the aerofoil's sectional properties, and the deflection curve y is obtained by integrating $\dfrac{(l-x)^2}{EI}$ twice, using the boundary conditions $y=\dfrac{dy}{dx}=0$ at $x=0$. The constant k in equation (3) could be chosen to make the dimensions consistent, but, as the method is essentially numerical rather than analytical, it is convenient to take $k=\dfrac{E}{10^4}$, which gives convenient numbers to work with.

ANALYSIS: ROTOR BLADE

For a rotor blade cantilevered at the disc the extra term for centrifugal potential energy is required:

$$\int_0^l\frac{A\rho}{2g}\Omega^2\left\{(x+R)\int_0^x\left(\frac{dy}{dx}\right)^2dx-y^2\cos^2\psi\right\}dx$$

The effect of twist is numerically small enough to justify the assumption of constant stagger ψ. The same assumption (3) is then made. When written in the form of a "centrifugal stiffening correction" to the static frequency ω_0, Rayleigh's method gives:

$$\frac{\omega_e^2-\omega_0^2}{\Omega^2}=\frac{\displaystyle\int_0^l\frac{A\rho}{g}(x+R)\int_0^x\left(\frac{dy}{dx}\right)^2dx\,dx}{\displaystyle\int_0^l\frac{A\rho}{g}y^2dx}-\cos^2\psi \quad (6)$$

In using this expression, the experimental value of ω_0 can be used if available.

For a rotor blade attached to the disc by a pin on which it is free to swing, the boundary condition $\frac{dy}{dx}=0$ at $x=0$ is replaced by $\frac{dy}{dx}$ is proportional to $EI\frac{d^2y}{dx^2}$ at $x=0$, in which the constant of proportionality is determined by the dimensions of the blade root and pin system. Equations (5) and (6) still hold.

COMPARISON WITH EXPERIMENTAL RESULTS AND OTHER METHODS OF CALCULATION

For making the comparison, a typical rotor blade was chosen, 3½ in. long. Four methods of calculation were used:—

(*a*) The standard method of finding any desired number of modes of a complex elastic system is to divide it into a set of discrete masses linked by light flexible rods, taking into account torsional-flexural coupling where necessary. The large set of equations of motion can then be solved by a standard matrix iteration procedure on a digital computer. In this case, five masses were used.

(*b*) The bending moment method (equation (3)) described in this note.

(*c*) Rayleigh's method, using equation (2).

(*d*) Rayleigh's method, using equation (1).

Figure 1 compares the deflection curves of the blade when rigidly clamped at the root, as calculated by the four methods. (*b*) agrees well with (*a*), presumably the most accurate. The frequencies calculated by these methods were:—

(*a*) 321 c./sec.
(*b*) 347 c./sec.
(*d*) 359 c./sec.

as against the measured mean of 13 blades, which was 334 c./sec. (316 minimum, 356 maximum).

For several other blades of different sizes, calculated and observed frequencies have been compared, and also the ratio

$$\frac{\text{stress shown by a strain gauge near the root fillet}}{\text{tip amplitude}}$$

(This quantity is of interest in the study of compressor blade vibration.) The calculated values fall within the limits of experimental scatter.

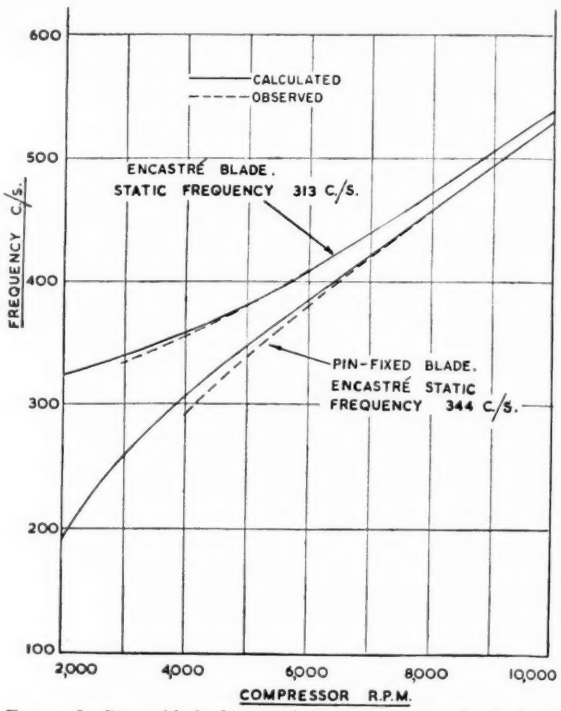


FIGURE 2. Rotor blade frequencies. Comparison of calculated and observed values.

The bending moment method has also been used to calculate the blade frequency while the engine is rotating, both with a fir-tree root and a pin root; measured values of ω_0 and the pin root dimensions were used. Fig. 2 compares the calculated values with experimental frequencies; the agreement is very good.

DISCUSSION

The matrix method is unnecessarily complicated and expensive to use for estimating the lowest natural frequency of an axial compressor blade. The uncertainty as to some of the sectional properties of an aerofoil used in the calculation, and the frequency variation of a set of blades manufactured to the same drawing within acceptable tolerances, limit the useful accuracy attainable by any calculation.

The advantage of the bending moment method described is that it can be used to make a rapid and reliable estimate of a blade frequency, and avoids the difficulty of graphical differentiation involved in using the conventional form of Rayleigh's method.

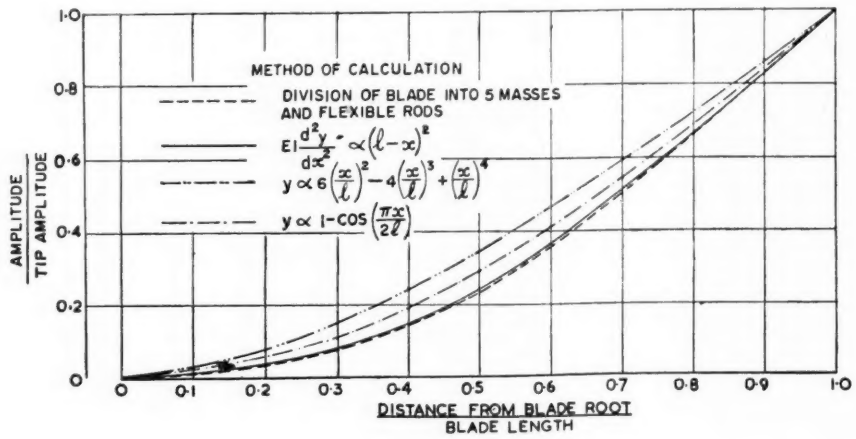


FIGURE 1. Comparison of the deflection curves of an encastred compressor blade obtained from different approximations.

The principle of assuming a bending moment curve can clearly find a much wider application than calculating blade frequencies. It can be extended to the same generality as the matrix method of dividing an elastic body into masses and rods, but in doing so the arithmetic becomes very similar to that required in the matrix method, and shows no advantage over it. A short account of this more general theory is given in the Appendix.

CONCLUSION

A simple way of estimating the lowest natural frequency of both rotor and stator blades has been described. It gives a value which, in practice, lies between the maximum and minimum observed frequencies of a set of the blades, when manufactured. The stresses and energies involved can also be estimated. The method has been found of practical use in studying compressor blade vibration, and the principle of postulating a bending moment curve instead of a deflection curve has a general application to beam vibration theory.

APPENDIX

EXTENSION TO HIGHER MODES

The success of the assumption that the blade vibrates only in the plane of the major principal axis depends on the thinness of the blade. For a turbine blade, especially if it is shrouded, and for a more general shape of cantilever, deflections in other directions, and twisting, must be taken into account. For higher modes this extension is also necessary, even for a thin blade.

The equations of motion of the blade will now be obtained. At any section x from the encastred end, take a set of axes through the undeflected flexural centre parallel to the undeflected principal axes (Fig. 3).

The elastic equations of the section are:

$$\left.\begin{aligned} \frac{\partial^2}{\partial x^2}\left(EI_y\frac{\partial^2 y}{\partial x^2}\right) &= w_y \\ \frac{\partial^2}{\partial x^2}\left(EI_z\frac{\partial^2 z}{\partial x^2}\right) &= w_z \\ \frac{\partial}{\partial x}\left(GK\frac{\partial\theta}{\partial x}\right) &= -M_F \end{aligned}\right\} \quad . \quad . \quad (7)$$

The inertia loads are:

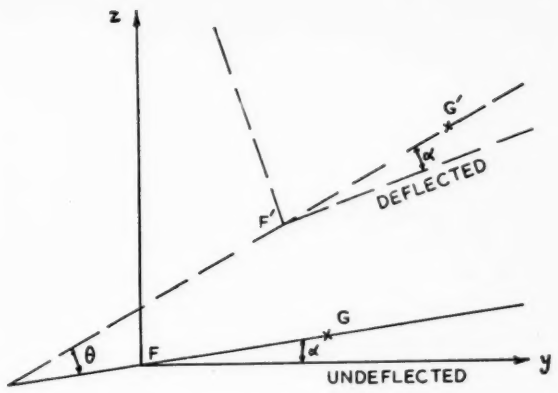
$$\left.\begin{aligned} w_y &= -\frac{A\rho}{g}\frac{\partial^2\xi}{\partial t^2} \\ w_z &= -\frac{A\rho}{g}\frac{\partial^2\eta}{\partial t^2} \\ M_G &= -\frac{I_p\rho}{g}\frac{\partial^2\theta}{\partial t^2} \end{aligned}\right\} \quad . \quad . \quad . \quad (8)$$

By geometry,

$$\xi = y + c\cos(\alpha+\theta)$$
$$\eta = z + c\sin(\alpha+\theta)$$

So, making the usual assumptions for a small oscillation that θ and $\frac{\partial\theta}{\partial t}$ are small,

$$\left.\begin{aligned} \frac{\partial^2\xi}{\partial t^2} &= \frac{\partial^2 y}{\partial t^2} - c\sin\alpha\frac{\partial^2\theta}{\partial t^2} \\ \frac{\partial^2\eta}{\partial t^2} &= \frac{\partial^2 z}{\partial t^2} + c\cos\alpha\frac{\partial^2\theta}{\partial t^2} \end{aligned}\right\} \quad . \quad . \quad (9)$$



F Flexural centre, $FG=c$. $F'(x, y)$ Deflected flexural centre.
G Centroid. $G'(\xi, \eta)$ Deflected centroid.

FIGURE 3.

Taking moments about F,

$$M_F = M_G - w_y c\sin\alpha + w_z c\cos\alpha \quad . \quad (10)$$

Apply (9) and (10) to (7) and (8). Assume that the deflections and the twist of the blade are sinusoidal, and of frequency ω_0. Then writing y, z and θ as amplitudes, the frequency equations are:

$$\left.\begin{aligned} \frac{d^2}{dx^2}\left(EI_y\frac{d^2 y}{dx^2}\right) &= \frac{A\rho}{g}\omega_0^2(y - Z\theta) \\ \frac{d^2}{dx^2}\left(EI_z\frac{d^2 z}{dx^2}\right) &= \frac{A\rho}{g}\omega_0^2(z - Y\theta) \\ -\frac{d}{dx}\left(GK\frac{d\theta}{dx}\right) &= \frac{I_p\rho}{g} + \frac{A\rho}{g}(Y^2+Z^2)\omega_0^2 - \\ &\quad - \frac{A\rho}{g}\omega_0^2(Zy + Yz) \end{aligned}\right\} \quad (11)$$

For the case $z=0$, these equations reduce to those given on p. 409 of Ref. 1. As the directions of y and z are arbitrary, Y and Z can be taken as positive.

These equations can be solved numerically by the Ritz method, originally given in Ref. 2, and described in Articles 61 and 62 of Ref. 1.

$$\left.\begin{aligned} y &= a_1 y_1 + a_2 y_2 + \ldots . a_i y_i \ldots . \text{ to } n_1 \text{ terms}, \\ z &= b_1 z_1 + b_2 z_2 + \ldots . b_i z_i \ldots . \text{ to } n_2 \text{ terms}, \\ \theta &= e_1\theta_1 + e_2\theta_2 + \ldots . e_i\theta_i \ldots . \text{ to } n_3 \text{ terms}. \end{aligned}\right\} \quad . \quad (12)$$

where the a's, b's and e's are unknown constants, and the individual terms y_i, z_i and θ_i are obtained by assuming:

$$EI_y\frac{d^2 y_i}{dx^2} = (l-x)^{1+i}$$

$$EI_z\frac{d^2 z_i}{dx^2} = (l-x)^{1+i}$$

but $GK\frac{d\theta_i}{dx} = (l-x)^i$, and integrating twice, in the same way as in the simple case. The number of terms to take depends on how complex a mode is to be investigated. For calculating the three lowest modes of a shrouded turbine blade, $n_1=2$, $n_2=1$, and $n_3=2$ should suffice.

The Ritz method (sometimes called the Galerkin method) gives the set of equations:

$$\left.\begin{aligned}
&\int_0^l \frac{d^2}{dx^2}\left(EI_y\frac{d^2y}{dx^2}\right)y_i dx=\omega_0^2\int_0^l\frac{A\rho}{g}(y-Z\theta)\,y_i dx\\
&\qquad\text{for each individual value of } i,\\
&\int_0^l \frac{d^2}{dx^2}\left(EI_z\frac{d^2z}{dx^2}\right)z_i dx=\omega_0^2\int_0^l\frac{A\rho}{g}(z-Y\theta)\,z_i dx\\
&\qquad\text{for each individual value of } i, \text{ and}\\
&-\int_0^l\frac{d}{dx}\left(GK\frac{d\theta}{dx}\right)\theta_i dx=\omega_0^2\int_0^l\left[\frac{I_p\rho}{g}+\right.\\
&\left.+\frac{A\rho}{g}(Y^2+Z^2)\,\theta\theta_i-\frac{A\rho}{g}(Z_y+Y_z)\,\theta_i\right]dx\\
&\qquad\text{for each individual value of } i.
\end{aligned}\right\}\quad(13)$$

These are $(n_1+n_2+n_3)$ simultaneous linear equations for the $(n_1+n_2+n_3)$ constants a, b and e. The numerical values of the coefficients can be obtained by graphical integrations. The frequency equation is, as usual, the $(n_1+n_2+n_3)$ order determinant eliminating all the constants. The modal shape can be found by evaluating the constants after finding the frequency.

For modes above perhaps the fourth, the accuracy will be poor, because shear deflection and rotary inertia have been neglected.

If $z=\theta=0$, and only a single term of y is taken, the single equation remains from (13):

$$\int_0^l \frac{d^2}{dx^2}\left(EI_y\frac{d^2y}{dx^2}\right)y\,dx=\omega_0^2\int_0^l\frac{A\rho}{g}y^2dx \quad . \quad (14)$$

which is, of course, the equation used to obtain the frequency approximation for the fundamental mode of a stator blade, equation (5).

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2. RITZ, W. (1911). *Gesammelte Werke*, Paris, 1911.

A Standardised Method of Representing Fatigue Test Results

by

J. C. SIMMONS, M.Sc.(Eng.), Grad.R.Ae.S.
(Technical Department, Royal Aeronautical Society)

IN HIS note in the June 1958 JOURNAL (p. 456) M. Fine outlines a new method of presenting fatigue test results and he suggests that this method results in greater accuracy of fatigue life estimation, particularly in the flat region of the *S-N* curve. Before this suggestion is accepted, more detailed attention should be given to a number of points.

If, first, we accept Fig. 1 of Fine's note as an exact representation of the fatigue characteristics of the material under some particular type of loading, then we can accept the presentation of the data as given in Fine's Fig. 2. The difficulty in reading from Fig. 1 occurs at values of N greater than, say, 10^7 and thus this is the region of interest. This region is found to be completely contained in the small square next to the origin in Fig. 2, and thus little increase in scale has been achieved. It is, of course, possible to enlarge the scale of this small square as is shown in Fig. *A* of this note. For clarity, lines for $p=0$ and $p=14{,}000$ lb./in.^{2} only have been included. It will be observed that lines of constant N may be drawn on this figure to give a clearer indication of the significance of the presentation.

Even with the enlarged scale of Fig. *A*, however, sample calculations show that final error due to a combination of:

(*a*) the error in reading the curves of Fig. 1 in order to determine the curves of Fig. 2 or Fig. *A*,

(*b*) the error in reading $\bar{S}$ from Fig. 2 in order to calculate X, and

(*c*) the error in reading from the curves of Fig. *A*,

is likely to be of the same order as the errors introduced by different people in reading from Fig. 1.

A more fundamental objection, however, arises from the fact that no estimate of fatigue life can be more precise than the data on which this estimate is based. With fatigue tests, scatter arises from a number of sources, few of which can be entirely eliminated and, even if specimens could be prepared which were perfectly identical in all respects, scatter would still arise due to various features of the testing machine. It is, for example, very difficult

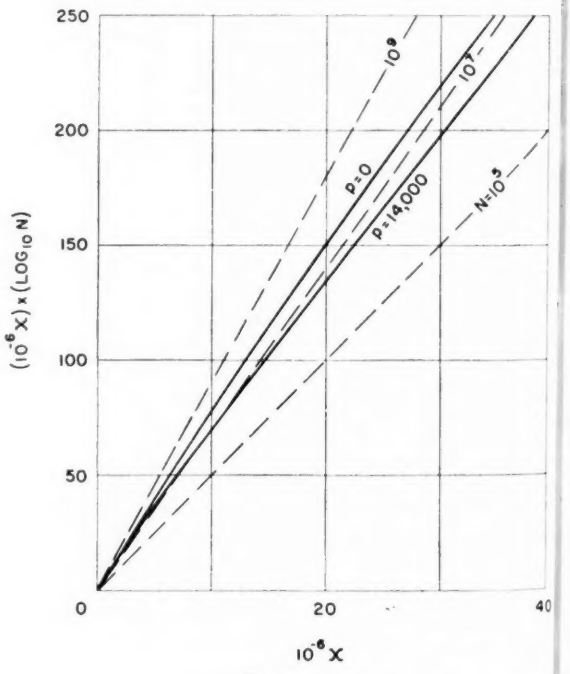


FIGURE *A*.

Received 16th July 1958.

to calibrate a fatigue machine under dynamic conditions and it is doubtful if the vibratory load actually applied to a specimen is known to an accuracy better than ± 5 per cent.

For tests on simple specimens under laboratory conditions, the fatigue life at a given stress level may vary by a factor of as much as 10 and the scatter is usually greatest at the flat end of the *S-N* curve. Thus, however many test results the *S-N* curve is based on, fatigue life estimation of any specimen or component can never be very precise and there is little point in presenting the data in other than the simplest form. This lack of precision must be particularly borne in mind if, as mentioned by Fine, a formula is used to represent the curves of Fig. 2, since the use of an empirical equation always tends to obscure the basic accuracy of the original data.

Implicit faith in the *S-N* curve at the high fatigue life end is, in any case, unnecessary since, due to the flattening of the curves, a few per cent reduction in design stress will give a much enhanced fatigue life.

In conclusion it could be suggested that if greater consistency of reading is required, then the simplest course is to replot the flat end of the *S-N* curves to an increased vertical scale.

Reply by M. Fine

The aim of the note was consistency in comparisons of fatigue life, not increased accuracy in estimation; it might be needed in deciding whether estimated fatigue lives of two designs for similar members, submitted by different people, are due to differences in reading off curves or to differences in design. Hence almost all Mr. Simmons' comments, except the last paragraph, are not relevant to the purpose of the note; if combined discrepancies due to (*b*) and (*c*)—but not (*a*)—are not less than those due to different people reading from a curve such as Fig. 1 the note has failed in its purpose. If they are, and if the purpose is worthwhile, the question is how best to achieve it. The curves of Fig. 2 are so simple compared with those of Fig. 1 that they seemed worth publishing at least; and surely the criterion is ease in using curves rather than ease in preparing them.

In addition (*i*) I agree with most of the earlier comments made and hence (*ii*) any method of standardising such results should be confined to predicting differences in fatigue life due to differences in design not to predicting absolute values; (*iii*) all the curves in Fig. 2 should, theoretically, have a vertical tangent at the origin.

Received 14*th August* 1958.

Graduates' and Students' Section

Visit to Coopers

A visit to Cooper Cars at Surbiton has been arranged for October on a Saturday morning, but the date has not yet been settled. All those interested should immediately contact the Hon. Visits Secretary, Mr. A. R. M. Pickering, who will let them know final details. His address is "Riverside," Vicarage Walk, Bray, Berks.

New Officers

The officers of the Committee for 1958/59 are:

Chairman	J. R. Cownie
Hon. Secretary	P. D. Stewart
Hon. Treasurer	M. P. Laker
Hon. Lectures Secretary	G. E. Deadman
Hon. Visits Secretary	A. R. M. Pickering
Hon. Editor	P. T. Ross.

Representatives

The Glasgow Branch of the Royal Aeronautical Society has formed a Graduates' and Students' Section with Mr. G. H. K. Goold of 27 Clunies Road, Moss Park, Glasgow W2, as Secretary.

At Coventry, Mr. M. J. Goodyear, D.C.Ae. of Armstrong Siddeley, has agreed to represent us in this firm. His address is 16 Brighton Street, Stoke, Coventry.

The Autumn Lectures

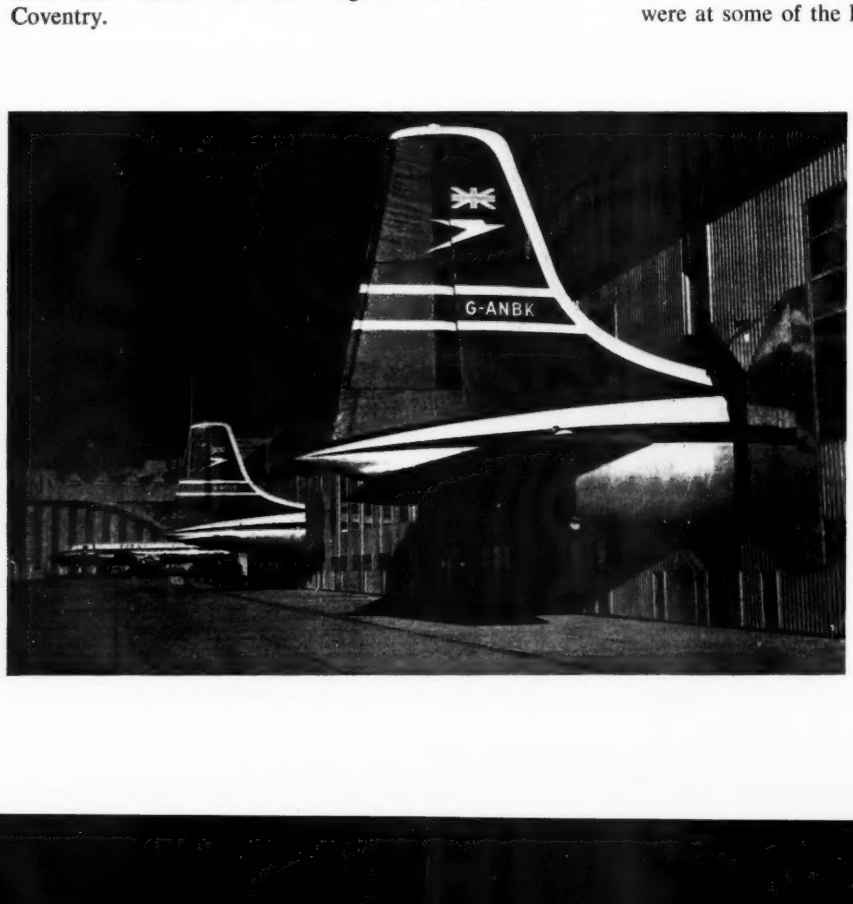
Enclosed with this issue of the JOURNAL is a card giving the Lecture and Social programme for this coming Autumn. There are six events altogether, comprising four lectures, the annual film show and the Winter Dance.

The opening lecture of the Session is to be held on 25th September when Mr. G. H. Lee of Handley Page will talk on "The Next Generation of Civil Aircraft." Mr. Lee's experience with large, high speed aircraft is extensive indeed and includes a great deal of work with the Victor bomber. His talk should therefore be both interesting and informative and channel current speculation on supersonic air travel.

The second lecture, on 16th October, is being given by Major P. L. Teed of Vickers-Armstrongs who will talk on "The Fatigue of Aircraft." Major Teed is one of the country's leading experts in this field and those attending can be assured of a well spent evening.

Both these lectures will begin at 7.30 p.m. in the Library at 4 Hamilton Place. They and all future lectures and film shows will be preceded by free coffee which will be served from 7 p.m. onwards. It is hoped that as many members as possible will avail themselves of this opportunity to meet other members in a social atmosphere and discuss matters of mutual interest.

The Committee hope that the Lecture Programme they have arranged will meet with your approval and that attendances will be considerably better than they were at some of the lectures earlier in the year.



Courtesy of B.O.A.C.

A view of the B.O.A.C. Headquarters and Engineering Base at London Airport, showing the new Wing Hangar in the foreground—scene of a recent visit by the Section.

THE LIBRARY

Reviews

PROPERTIES OF WROUGHT AND CAST ALUMINIUM AND MAGNESIUM ALLOYS AT ROOM AND ELEVATED TEMPERATURES. *P. H. Frith. H.M.S.O. for Ministry of Supply, London, 1958. 178 pp. Illustrated. 22s. 6d.*

Mr. P. H. Frith, with whose excellent experimental work metallurgists and aeronautical engineers have long been gratefully familiar, has, under the contractual inspiration of the Ministry of Supply, produced a stupendous work. It contains 178 foolscap pages, 93 tables and 19 figures. It weighs 1·8 pounds and, it has to be admitted, is very heavy reading indeed. The opus, and it is an opus and very definitely not an opuscule, deals, *inter alia*, with the fatigue, creep, tensile, Izod, thermal expansion, thermal conductivity, density and hardness, from room temperature to 350°C, of a large number of light alloys—most have B.S., or D.T.D. specifications.

The writer who has read through this tome has to confess that he did not at once feel the better for it. He, in fact, was suffering from factual fatigue. This is, however, not fair criticism. The theologian who successively and conscientiously perused each page of Cruden's Concordance, the author who did the same thing with the Oxford dictionary or the commercial traveller who made a similar study of Bradshaw would feel the same mental satiation. Each would nevertheless extol the sterling worth of their respective book, but would rightly contend it was not intended for reading but for reference.

Judged as it should be judged, namely, as a work of reference, Mr. Frith is entitled to very considerable praise, but also to some criticism. He has got together an immense amount of factual information, derived over a period of more than twenty years. For this he is undoubtedly deserving of the warm gratitude of a relatively small yet important public, for, unfortunately, not very many are interested in what has so profoundly interested him. He should nevertheless be criticised for failing to provide a work of this character with an index. His discussion of results is somewhat disappointing. One, however, appreciates that were he to attempt what would be so valuable, namely, an evaluation of the many different alloys, he would thereafter know no peace.

Lesser criticisms could be indulged in—one regrets no reference to the relative scatter in results which is considerably greater with some alloys than with others. In the case of the fatigue tests, the method whereby stress concentration is determined is not disclosed. There is, however, enough information to allow of the energetic reader working this out for himself. No warning is given as to the present chaotic state of knowledge with regard to stress concentration and notch sensitivity. Further similar criticisms of about the same magnitude could be mentioned, but this writer's last words should be to point out, what will no doubt be appreciated by many readers and particularly by those from across the Border, that, at 22s. 6d. per set, facts have seldom been supplied at a cheaper unit rate.—P. L. TEED.

VISTAS IN ASTRONAUTICS. *Morton Alperin, et al. Pergamon Press, New York, 1958. 330 pp. Illustrated. 105s.*

It is a curious fact that although British scientists and engineers have been to the forefront in all major advances (T.V., Radar, Nuclear Power and Jet Propulsion, to name but a few), our Parliament has hardly discussed our abstention from Astronautics. Little is known of Russian thought, only of their end product, the Sputniks; this book is essential reading for those who are interested in the vast American effort in Astronautics recently.

In February 1957, the first annual symposium on Astronautics was convened by the American Air Force Office of Scientific Research and Convair Aircraft. "Vistas in Astronautics" is a compendium of the papers presented at this conference. It shows all too clearly the difference in effort in the United Kingdom and U.S.A. on this subject; the book is full of actual measurements and photographs of apparatus in existence in the States some two years ago.

Forty-three papers are reprinted, and make a very readable volume broken down into six sections as follows:

I. Re-entry. II. Tracking and Communications. III. Environment and Measurements. IV. Propulsion. V. Orbits. VI. Human Factors.

In the Re-entry section it is refreshing to note consideration of atmospheres other than our own, and that "re"-entry into Martian or Venusian atmospheres should produce less kinetic heating than re-entry into our own atmosphere.

Only one of the re-entry papers considers the spherical shape seriously; the others postulate lift/drag ratios varying between 1 and 5. The "ice-skate" aerofoil sectioned delta proposed by Ferri, Feldman and Daskin looks the most promising shape.

Two of the tracking and communication papers tackle the problems in great detail; this section should be studied by the British electronics industry, as the sale of ground radar and satellite tracking stations must already be big business in Russia and U.S.A.

Environment and measurements deals with meteor collision risk (Whipple), upper atmosphere research, solar activity and radiations, cosmic rays, vehicle equilibrium temperature and the construction on earth of an inhabited high vacuum laboratory.

Propulsion deals not only with conventional fuels but plasma jets, ionic propulsion and nuclear energy. It would appear that security has prevented the description of current U.S. rocket motors.

Orbits (V) starts with Oberth's 40-page proposal to use a pendulum to control the attitude of a satellite, and continues with other papers on accuracy requirements. The last section on "Human factors" deals with housekeeping problems, cosmic-ray hazards and practical experiments on men confined under laboratory simulated space conditions, and ends with a legal paper.

All in all a good book, excellently printed, well worth £5 5s. 0d. and a timely reminder that space-flight is not around the corner, but an everyday occurrence, and that we in Britain should arouse ourselves to ask why we spend millions of pounds on aircraft, missiles, nuclear power, and so on, and not even hundreds of pounds on space-flight.—W. F. HILTON.

PROCEEDINGS OF THE INTERNATIONAL CONFERENCE ON FATIGUE OF METALS, 1956. *Arranged and published by the Institution of Mechanical Engineers with the co-operation of the American Society of Mechanical Engineers, 1958. 968 pp. Illustrated. 90s.*

No fatigue worker or stress analyst can afford to be without this book if he wants to be up-to-date. It is a record of a Conference on fatigue sponsored by the

I.Mech.E. in co-operation with the A.S.M.E., held in London in September 1956, with a follow-up Conference in New York in November of the same year. Although it may seem that this was a joint Anglo-American affair, the Conference was actually truly international, as from a total of eighty-one carefully selected papers presented, six were Russian, five German, three French, three Australian, and two Japanese. In addition, there were quite a number of other nationals among those who participated in the Discussions. Thus, as far as is at all possible, this book represents the present state of knowledge on the subject with all its manifold ramifications.

The merits of any publication must, of course, be judged against the background of the contemporary stream of published information, which in the case of fatigue has taken on staggering proportions. Particularly, the years 1955-56 were extremely productive, and, in fact, two notable international conferences preceded this Conference; the first was at Stockholm, in May 1955, known as the IUTAM-Colloquium, the second was held at Columbia University, in January 1956, on "Fatigue in Aircraft Structures." The proceedings of these two conferences, taken together, produced 810 pages of standard book size, against 968 pages demi-quarto of the present Proceedings which, at fifty-three per cent of the joint price of the two other books is extremely good value. As to the authors, 13 of them (16 per cent) were also contributors to either the Stockholm or the Columbia conferences, but this should not cast doubt on the originality of their contributions. New facts need frequent re-formulation before they sink in, and therefore a certain amount of repetition is healthy. In fact, in the reviewer's view, one of the most profound statements contained in the Proceedings refers to the fact that there is a discrepancy between what we already know about fatigue and the amount of knowledge we actually manage to apply. Where the Proceedings score heavily is in the manner in which the individual papers were introduced, summarised and commented upon by eight British and eleven American Reporters, each of whom is a well-known authority in his special field. Of equal importance are the Discussions which were taking place both in London and New York, Communications and Authors' Replies, to which 190 most useful pages are devoted. Plenary Sessions were held, again in London and New York, in which the Reporters summed up the position as it emerged from the Discussions.

Indeed, from among the mass of publications, the present Proceedings stand out as a landmark for authoritative guidance and future reference. However, this does not mean that there was agreement on all problems. Far from it, as just one example should show: on page 803 it is stated that ". . . it was wasted effort to pursue any further the chimera of the relation between k_t and k_f. It leads nowhere." While on page 931 one can read: "This problem of the relationship between k_t and k_f was generally considered to be one of the most important problems in fatigue today, both from the theoretical and practical standpoints."

From the aeronautical engineer's point of view, ten of the papers are of direct importance, while at least fifty per cent of the remaining papers are of relevance. An interesting and much-discussed paper is on fail-safe design, but unfortunately the author's reply is not forthcoming. There is one paper of major importance on statistics, and several papers by well-known British authors. The stress analyst will find several papers of great interest on combined stresses in relation to fatigue. Perhaps the most spectacular papers are two Australian ones, one on "Random Noise Fatigue Testing" and the other on full-scale fatigue tests on ninety-one P-51-D Mustang fighter wings. Spectrum loading is represented by a further two papers (one American, one German), and is widely discussed and commented on, particularly in the U.S.A., where the general consensus of opinion appears to be that it will always be inadequate to use methods of life estimation based on constant amplitude test results as these cannot account either for cumulative damage, or for the actual failure location of service. When will this knowledge be applied?—T. HAAS.

MATHEMATICAL THEORY OF COMPRESSIBLE FLUID FLOW. *Richard von Mises. Academic Press, New York, 1958. 514 pp. Illustrated. 15 dollars.*

This is the third volume of a series of monographs on Applied Mathematics and Mechanics which started with Oswatitsch's "Gas Dynamics." Professor von Mises died before completing this book; the co-authors, Hilda Geiringer and G. S. S. Ludford, are to be complimented on bringing the work to fruition and maintaining continuity of outlook throughout the book.

The first two words in the title must be stressed; this is a book dealing with some of the more fundamental principles and researches of compressible fluid flow from a mathematical point of view. The presentation differs considerably from that in most modern works in this field in which the physical point of view is emphasised. In some cases, for example, in showing that the energy equation is a mathematical consequence of Newton's equation and the equation of continuity, there is a lot to be said for this approach. The need of a specifying equation relating, for example, pressure and density, is well brought out and shows the role of thermodynamics in compressible flow theory.

The loss of the physical picture is, however, felt acutely in other parts of the book. So often we see things as through a hodograph (possibly because of its mathematical interest), whereas a few well chosen words and diagrams would help the reader to grasp the practical importance of the mathematics. For example, the theory of simple waves is not simple; if some mention of their physical application could have been made much earlier in this section (Article 18), one reader would have been much happier. Insufficient emphasis would appear to be given to the important Prandtl-Meyer expansion and to flow through a convergent-divergent nozzle (this is not mentioned explicitly until the last chapter of the book).

The lack of a physical point of view is very evident in the section on oblique shock waves, notably on the occurrence of a weak or strong shock.

One dimensional flow, the theory of characteristics, plane steady potential flow and shock theory are treated very fully and, in the final article, some interesting mathematical problems dealing with transonic flow are discussed. Many aspects of the subject are, however, not treated at all—in particular, linearised theory and flow over wings.

The book will be of most use to research workers with a mathematical bias wishing to do further research in a specialised part of the field; such readers will find in the notes and addenda at the end of the book a comprehensive list of recent papers.—A. W. BABISTER.

NO ECHO IN THE SKY. *Harald Penrose. Cassell, London, 1958. 134 pp. Illustrated. 15s.*

Surely every airman is aware of beauty but few who write of every-day flying try to put the wonder of their feelings into poetical prose. For those who aspire to do

so, this is a book to study reflectively, whether they would write to express themselves, to communicate to landsmen what only airmen know, or to evoke echoes in the minds of fellow airmen.

Trained as an engineer, Harald Penrose, for twenty-five years a test pilot with The Westland Aircraft Company, knows that poetry is the right medium for describing the airman's world and work. It is not his fault that the present state of the art of powered aviation and the technical pre-occupations of a test pilot are such that gravity is more easily overcome than bathos. He is at his best when describing a flight in his own, home-made sailplane over and along the coast of Dorset: Chapter VIII —Soaring Sea Quest. I commend questing landsmen to start with that chapter, and I believe that they will then want to go to the beginning and fly in sympathy with this sensitive aviator through his adventurous career in English skies and over English countryside. And all the while they will be reminded of that central wonder and assurance in the airman's personal faith:—

> "The quiet curiosity and conjecture of what may happen in the unknown of any flying venture has no apprehension. Instead, one's insignificance in the vast amphitheatre of the sky seems to draw a power beyond mortality, which restores the vision of the earth to greater brilliance."
>
> "I was suddenly closer to an understanding of the inexplicable. I knew that my life, and the wings on which I was poised, and the skies and the earth, were integrated in some profound design."

Such sentences appear in every chapter. Alas, too often the "works" decline to chime with the exalted mood of the flier:

> "In a realm so high and vast and empty that it seemed the outskirts of eternity, I flew . . . dreaming my secret thoughts in a life maintained by an artificial stream of oxygen, and a pressurisation system that warmed and compressed the cabin air to conditions equivalent to a relative height of 22,000 ft. less than the true altitude. Perhaps because of its strangeness the hiss of the pressure pump struck louder than the rough growl of the wing engines."

Perhaps you can take that quite happily. But can you take this?—

> "The future breathed promise that there would be other days and other years, more and yet more, to gather to my heart."

This is a peaceful, courageous, sincere book and a pleasant one. If at times one wonders whether it really *is* sincere, or whether some of the fancies expressed were not conjured-up for writing or borrowed from elsewhere, the explanation of that doubt may be that the author *aspires* to poetry and unconsciously uses sets of words that one vaguely recognises ("The breeze whispered exciting wordless promises that set veins tingling") or that make one mutter, "cliché!" It is a book that most airmen would blush to write: it discloses their secret thoughts, always fuller of humility than most landsmen suppose; thoughts that most airmen forbear to uncover because they belong to the feminine part of a man's nature.

Keith Shackleton's headpieces in linocut make him, too, seem to be an engineer-draughtsman turned airman-poet. They show a fine sympathy with the narrative—sending an echo to the sky.—SIR VICTOR GODDARD.

THE EXTERIOR BALLISTICS OF ROCKETS. *Leverett Davies, Jr., James W. Follin, Jr. and Leon Blitzer. Van Nostrand, New Jersey, 1958. 457 pp. Illustrated. 64s.*

A wide gulf divides the language of aerodynamics from that of ballistics, and the two sciences have in the past advanced along different lines. The motion of unguided rockets is a subject which bridges the gulf however; quadrant elevation co-exists with climb angle, resistance with drag, and cross-force with lift. The authors of this book have feet firmly planted in both camps, and they provide a most satisfactory exposition of the mathematical theory of the motion of unguided rockets, both fin-stabilised and spin-stabilised. The dispersions arising through thrust misalignment, gravity, wind, and so on, are fully discussed, for both ground-launched and air-launched missiles.

The chief differences between the theory of unguided rockets and the theory of the motion of aircraft or guided missiles arise because thrust forces are dominant while the rocket is burning, because constant-speed approximations fail in face of such high acceleration, because there are no moving control surfaces, and because the missile is usually assumed to be axially symmetrical. Rudimentary aerodynamics suffices for the unguided rocket, and some of the remarks about drag calculation in this book may make the aerodynamicist shudder.

Though little is said in the book about wider applications, the theory can readily be extended to cover the unguided boost phase of guided missiles and the later stages of propulsion in a satellite launching missile like the Jupiter-Explorer. The application of the theory in this wider realm is limited however by the assumptions that the rocket is axially symmetrical and that its mass and acceleration are constant. The constant-mass approximation may be tolerable when the weight of propellant is only 20 per cent of the total, but not when it is 70 per cent. And missiles, though they often have a plane of symmetry, do not usually possess axial symmetry.

The book is the fruit of the theoretical work on exterior ballistics in the U.S.A. during the years 1942-8. The final report on the work was issued by the U.S. Office of Naval Research five years ago, and, after certain deletions and corrections, has been reprinted to form this book, an impressive volume which is likely to remain a standard work on the subject for many years. The book is handsomely produced and clearly printed, and it cannot be called unduly expensive.—D. G. KING-HELE.

LEADER OF THE FEW: The authorised biography of Air Chief Marshal, The Lord Dowding of Bentley Priory, G.C.B., G.C.V.O. *Basil Collier. Jarrolds, London, 1957. 264 pp. Illustrated. 25s.*

Lord Dowding will always be remembered as the Leader of the famous "Few" who so decisively prevented an invasion of the United Kingdom in the late summer of 1940. The series of actions which came to be called the Battle of Britain succeeded because, first, Fighter Command was prepared and waiting and, secondly, because the Country obstinately refused to accept defeat. The fact that we were ready and waiting was largely due to the stand taken by Sir Hugh Dowding, then Commander-in-Chief, Fighter Command, who had persistently demanded the necessary fighter aircraft and radar equipment to meet the attack which he knew was bound to come.

To Dowding must also go the credit for the introduction into the Royal Air Force of all-metal aircraft structures, of bullet-proof windscreens and, following upon the use of armourplate protection for pilots and vital

equipment by the Luftwaffe, of the 20 mm. Hispano Cannon. Radar itself was introduced into the defensive pattern of Fighter Command under the direction of Dowding. The Country was therefore blessed with a Commander-in-Chief of a vital defence weapon notable both for his vision of what would be needed to meet and to defeat the enemy's next move and for his stubborn determination to have his own way when he was convinced that he was right. He staked his reputation and his career on his estimation of the minimum number of fighter squadrons which would be needed after the fall of France and, in the event, this estimate proved to be accurate.

His obstinate refusal, in the face of pressure from the highest political level, to bleed increasing numbers of new Hurricanes away from the life-blood of home defence in support of a failing French army is something to which we may well owe our very existence as a sovereign state today. The fact that he was by-passed for the post of Chief of the Air Staff was a heavy blow to him but his loss was the Country's gain since, unfettered by political considerations inseparable from such an appointment, he was able forcibly to express his views on the need to preserve and as quickly as possible to build up the strength of Fighter Command.

Basil Collier is clearly a dogged biographer, and he must have dug more information out of Lord Dowding and his family than anyone ever before, but he has left the reader with a picture of this distinguished (if reputedly "difficult") officer which is far from distinct. The title "Leader of the Few" is rather unfortunate because although it smacks of journalism rather than of authorship, the "human angle" is sometimes hard to find. One can sympathise with the author because the subject is not an easy one. He has, however, gone to a great deal of trouble to delve into Dowding's early life, both as a child and as a young Gunner officer, with the intention of throwing a clearer light on to the creation of this enigmatic and dour personality. But, despite the painstaking presentation of fact and the careful blending of Dowding's personal background with those facts of history which can be published from official records, the enigma remains. Dowding, the man, is still almost unknown, although he is clearly far more human than those who served under him in 1940 believed him to be.

Collier has painted an accurate, if sometimes pedantic, picture and has clearly portrayed his subject's great integrity of purpose. He shows how this single-mindedness pervades all Dowding's actions, no matter in what direction his inclinations drive him, whether they be as a martial leader of men or, out of curiosity, as an explorer into the occult.

If Collier has failed at all, it is a failure to present Dowding as a human being, subject to human emotions but, unlike many of his fellows, overcoming those emotions by the conscious substitution of something which he believes to be most worthwhile.—A.S.C.L.

Additions to the Library

Aviation Facts and Figures 1958. Aircraft Industries Association, Washington. American Aviation Publications. 1958. 132 pp. $1.50. Although the price has increased by 50 cents it is still excellent value for money. As has been said in previous years, it contains tables of significant figures such as yearly outputs, manpower employments, accident statistics and so on, with the accent, naturally, on American production. Tables are supplemented by background articles. The chapter on missiles is quite frightening when one rates the expenditures made. 40 different types are listed.

Flight. A Pictorial History of Aviation. Year Inc., New York. 1958. 248 pp. $10. (Obtainable from John Calder Ltd., 17 Sackville Street, W.1.) The type of book to keep by one's bedside for dipping into. A collection of captioned photographs, it starts as near the beginning as is possible (man's dream of flight) and ends with "Conquest of Space." If you *do* keep it by your bedside your son will grab it every morning and you will not get him out of bed. Over 1,000 pictures take time to absorb.

Handbook of Layout and Dimensioning for Production. H. H. Katz. Macmillan, New York. 1957. 469 pp. Illustrated. £5 5s. 0d. *To be reviewed.*

Mechanics Applied to Engineering. G. H. Ryder. Cleaver-Hume, London. 1958. 242 pp. Diagrams. 21s. A short course in statics and dynamics for first year degree or diploma work, applying the basic principles to as many fields of engineering as possible; it includes worked examples and problems.

Notes on Analog-Digital Conversion Techniques. A. K. Susskind (*Editor*). Chapman and Hall, London. 1958. 421 pp. 80s. *To be reviewed.*

Space Exploration. P. Moore. Cambridge University Press, London. Readers' Guides; 3rd Series. 1958. 36 pp. 3s. A reading list, published for the National Book League, with a short introduction to the literature.

Standard Aircraft Handbook. 2nd Edition. S. Leavell and S. Bungay (*Editors*). Aero Publishers Inc., Los Angeles. 1958. 160 pp. Illustrated. $2.75. The second edition of a reference and training manual for aircraft workers and mechanics in factories and maintenance companies; it includes the recently changed aluminium alloy designations.

Sound Pulses. F. G. Friedlander. Cambridge University Press, London. 1958. 198 pp. Diagrams. 40s. One of the Cambridge Monographs on Mechanics and Applied Mathematics, this book provides an up-to-date description of the theory of sound pulses and its recent developments. The individual chapters deal with the equations of motion, wave fronts and characteristics, geometrical acoustics and their application to reflection problems, and the diffraction of a pulse by a wedge, circular cylinder, sphere and other objects.

Werkstoff-Handbuch der deutschen Luftfahrt. Part I. Metallische Werkstoffe. Ministry of Defence, Germany. Beuth-Vertrieb GMBH. Koln. 1958. 191.50 DM complete. *To be reviewed.*

World's Sailplanes, The. K. G. Wilkinson *et al.* (*Editors*). Ostiv, Berne. 1958. 230 pp. 17s. 6d. British Gliding Association (selling agents). *To be reviewed.*

I.A.S. PREPRINTS

NATIONAL SUMMER MEETING July 1958

835 Booster Case Design for Hypersonic Vehicles. P. E. Grafton and E. F. Styer.

836 Comparison of Approximate Solutions of the Thermoelastic Problem of the Thick-Walled Tube. J. H. Baltrukonis.

837 Analysis and Direct Measurement of the Skin Friction of Uniformly Rough Surfaces at Supersonic Speeds. F. W. Fenter.

838 Improved Reliability in Air-Ground Communications. B. F. Miller.
839 Hypersonic Digital Control Systems. A. E. Cooper.
840 The Rocket-Engine Test Facility Sets the Pace. B. F. Rose, Jr.
841 Redesign of Ramjet Engine Ground Test Facilities. S. D. Fasken.
842 Practical Limitations on Orbit Determination. R. M. L. Baker, Jr.
843 The Handling Qualities of VTOL Aircraft at Low Speed. P. F. Girard.
844 The Problem of Obtaining High Lift-Drag Ratios at Supersonic Speeds. C. E. Brown and F. E. McLean.

Reports

AERODYNAMICS

Boundary Layer

Special bodies added on a wing to reduce shock-induced boundary layer separation at high subsonic speeds. R. T. Whitcomb. N.A.C.A. T.N. 4293. *June* 1958.
The basic test configuration had a relatively thick 35° swept-back wing of aspect ratio 7·05 mounted on a contoured fuselage.—(1.1.4.2).

An experimental study of the turbulent boundary layer on a shock-tube wall. P. B. Gooderum. N.A.C.A. T.N. 4243. *June* 1958.
Interferometric measurements were made of the density profiles of an unsteady turbulent boundary layer on the flat wall of a shock tube. The investigation included both subsonic and supersonic flow (Mach numbers of 0·50 and 1·77) with no pressure gradient and with heat transfer to a cold wall. Velocity profiles and average skin-friction coefficients were calculated. Effects on the velocity profile of surface roughness and flow length are examined.—(1.1.3.4×1.5.1).

Average properties of compressible laminar boundary layer on flat plate with unsteady flight velocity. F. K. Moore and S. Ostrach. N.A.C.A. Report 1325. 1957. (*Supersedes T.N.* 3886.)—(1.1.1.4×1.9.1).

Attenuation in a shock tube due to unsteady boundary layer action. H. Mirels. N.A.C.A. Report 1333. 1957. (*Supersedes T.N.* 3278.)
A method is presented for obtaining the attenuation of a shock wave in a shock tube due to the unsteady boundary layer along the shock tube walls. It is assumed that the boundary layer is thin relative to the tube diameter and induces one-dimensional longitudinal pressure waves whose strength is proportional to the vertical velocity at the edge of the boundary layer.—(1.1.1.4×1.12.1.3).

Étude expérimentale de la couche limite turbulente et de son intéraction avec l'onde de choc sur un demi-profil en ecoulement transsonique. R. Michel. O.N.E.R.A. N.T. 47. 1957. (*In French.*)
Experimental investigation of the turbulent boundary layer on an aerofoil in transonic flow for several shock and compression wave positions.—(1.1.2.3).

On the temperature distribution in thin flat plates with laminar supersonic boundary layers. M. Tideman. S.A.A.B. T.N. 39. *January* 1958.—(1.1.1.4).

Compressible Flow—*see also* Testing and Instruments

Supersonic flow past quasi-cylindrical bodies of almost circular cross-section. D. G. Randall. R. & M. No. 3067. 1958.—(1.2.3.1).

Methods for determining the wave drag on non-lifting wing-body combinations. L. M. Sheppard. R. & M. No. 3077. 1958.
The area-rule, moment-of-area-rule and transfer-rule methods for estimating the wave drag of wing-body combinations are discussed. Theoretical methods and their associated numerical techniques are given but no numerical results or particular applications.—(1.2.3.1).

A fuselage addition to increase drag-rise Mach number of subsonic airplanes at lifting conditions. R. T. Whitcomb. N.A.C.A. T.N. 4290. *June* 1958.
The addition of fuselage volume on the top of the forward portion of the fuselage for the purpose of increasing the drag-rise Mach number of subsonic aeroplanes at lifting conditions is investigated. The additions have been designed on the basis of the area rule and other important considerations to provide greater practicability of application compared with shapings previously investigated.—(1.2.1).

A numerical method for evaluating wave drag. M. S. Cahn and W. B. Olstad. N.A.C.A. T.N. 4258. *June* 1958.
A numerical method for evaluating von Kármán's wave-drag equation was developed and applied to four analytical bodies of revolution.—(1.2.3.2).

The interaction of a plane strong shock wave with a steady magnetic field. J. H. deLeeuw. U.T.I.A. Report 49. *March* 1958.
A theoretical investigation was made of some aspects of the interaction of a magnetic field and the flow of ionised and electrically conductive argon, as it may be produced in the shock tube by strong shock waves. Three different configurations were considered. The axial components of the ponderomotive forces that act on the gas as a result of its motion through the magnetic field were calculated for each of the configurations for shock tubes with non-conductive walls.—(1.2.4.2×1.12.1.3).

Control Surfaces—*see* Wings and Aerofoils

Internal Flow—*see also* Boundary Layer, Mathematics, Power Plants

Internal characteristics and performance of several jet deflectors at primary-nozzle pressure ratios up to 3·0. *J. G. McArdle. N.A.C.A. T.N.* 4264. *June* 1958.
Several model jet deflectors were tested in quiescent air. The deflectors (swiveled nozzles, auxiliary nozzles and mechanical deflectors) were designed to be adaptable to conventional turbojet exhaust systems. Analytical expressions relating the performance of each type deflector with significant design variables are also presented.—(1.5.1.2).

Use of the Coanda effect for obtaining jet deflection and lift with a single flat-plate deflection surface. U. H. von Glahn. N.A.C.A. T.N. 4272. *June* 1958.
Performance characteristics similar to those for a flap immersed in an air stream were obtained in quiescent air for nozzle-deflection-plate configurations using the Coanda effect for achieving jet deflection and vertical lift.—(1.5.1×1.8.2).

Stall propagation in a cascade of airfoils. A. R. Kriebel et al. N.A.C.A. T.N. 4134. *June* 1958.
An experimental investigation of stall propagation in a stationary circular cascade in which high-speed schlieren and interferometer photography is used is described.—(1.5.2.3).

Investigation of the effectiveness of a windmill for improving flow uniformity in a duct. J. S. Murphy and D. W. Clutter. Douglas Report ES 26393. *October* 1957.
Tests have been made to determine the effectiveness of a freely rotating windmill in smoothing out non-uniform duct flow. The design of a windmill and the forces on a windmill due to changes in wind speed are discussed. The results show the effect of blade angle and number of blades on the windmill's effectiveness in smoothing out a wide variety of disturbance profiles, both circumferential and radial.—(1.5.1×29.1).

Exact solution of the Neumann problem calculation of non-circulatory plane and axially symmetric flows about or within arbitrary boundaries. A. M. O. Smith and J. Pierce. Douglas Report ES 26988. *April* 1958.

A method has been developed and applied to the calculation of low-speed flows about or within bodies of almost any shape, provided the flow is either plane or has axial symmetry. Solid-body, inlet and purely internal flow problems can be solved. The method is capable of dealing with several bodies at once in the presence of one another, and consequently interference problems can be treated with ease. Boundaries need not be solid, that is, flows involving area suction can be calculated. Velocities can be computed not only for points on the surface of the body but for the entire flow field.—(1.5.1 × 1.10.1.1).

STABILITY AND CONTROL—*see* INTERNAL FLOW
WINGS AND AEROFOILS

THERMO-AERODYNAMICS—*see also* BOUNDARY LAYER
THERMODYNAMICS

An approximate analytical method for studying entry into planetary atmospheres. D. R. Chapman. N.A.C.A. T.N. 4276. *May* 1958.

The complete pair of motion equations for entry into an exponential atmosphere is reduced to an ordinary, non-linear differential equation of second order by introducing a mathematical transformation. This equation includes terms representing the gravity force, centrifugal force, lift force, drag force and vertical acceleration. Study is made of the decelerations, heating rates and total heat absorbed for entry into Venus, Earth, Mars and Jupiter.—(1.9).

Séminaire d'aérothermique de la faculté des sciences de Paris. E. A. Brun. Pubs. Sc. et Tech. N.T. 73. 1958. (*In French.*)—(1.9 × 30).

WINGS AND AEROFOILS—*see also* INTERNAL FLOW
MATHEMATICS

Exploratory wind tunnel investigation to determine the lift effects of blowing over flaps from nacelles mounted above the wing. J. M. Riebe and E. E. Davenport. N.A.C.A. T.N. 4298. *June* 1958.

An exploratory investigation, in which high-pressure air was blown over single and double slotted flaps from externally supported steel-pipe nacelles above the wing to simulate jet engine exhaust, has been conducted on a model with a delta wing of aspect ratio 3. The jet-momentum coefficient range was from 0 to 3·0 and the model angle of attack was 0°.—(1.10.2.2 × 1.3.4 × 1.8.2).

HELICOPTER AERODYNAMICS—*see also* STRUCTURES—LOADS

Normal component of induced velocity for entire field of a uniformly loaded lifting rotor with highly swept wake as determined by electromagnetic analog. W. Castles et al. N.A.C.A. T.N. 4238. *June* 1958.

Values of the normal component of induced velocity throughout the entire field of a uniformly loaded rotor at high speed are presented in the form of charts and tables. Many points were found by an electromagnetic analog, details of which are given.—(1.11.3).

TESTING AND INSTRUMENTS—*see also* COMPRESSIBLE FLOW
SCIENCE—GENERAL
MECHANICAL ENGINEERING

An experimental investigation of the flow properties behind strong shock waves in nitrogen. H. F. Waldron. U.T.I.A. Report 50. *March* 1958.

Pressure, particle velocity and speed of sound were measured in the uniform region behind shock waves up to a shock Mach number of 13. From the measured quantities, the pressure, speed of sound, density, temperature, enthalpy and flow Mach number were calculated. The attenuation of a shock wave formed by H_2-O_2-He mixtures was investigated experimentally for three methods of ignition and compared to the attenuation of the shock wave produced when cold hydrogen was used as the driver.—(1.12.1.3 × 1.2.3.2).

U.T.I.A. air duct facility for investigation of vibration noise induced by turbulent flow past a panel (boundary layer noise). L. Maestrello. U.T.I.A. T.N. 20. *April* 1958.

An acoustically-quieted air-duct facility has been constructed for the purpose of investigating noise generated by turbulent flow past a flexible panel. The facility is basically an open-circuit wind tunnel with interchangeable rectangular duct sections. The downstream end of the duct contains the test panel, fitted flush in one wall. Details of design and aerodynamic performance are given.—(1.12.1.1 × 32.2.3).

AEROELASTICITY

See also MATHEMATICS
STRUCTURES—LOADS
THEORY AND ANALYSIS

Flutter analysis of rectangular wings of very low aspect ratio. R. W. Fralich and J. M. Hedgepeth. N.A.C.A. T.N. 4245. *June* 1958.

The analysis uses slender-body aerodynamic theory and thin-plate theory. The spanwise variation of wing deflection is assumed to be given by a parabola, and the chordwise variation is allowed complete freedom. The results show the variation of flutter speed and mode shape with aspect ratio. Comparisons are made with additional results obtained by approximating the chordwise deflection shape by use of parabolic or cubic curves.—(2).

INSTRUMENTS AND EQUIPMENT

See also POWER PLANTS

Some factors influencing the speed of response of hydraulic position servomechanisms. C. W. Lewis. R. & M. No. 3089. 1958.

A study is made of the influence of working pressure, relay torque-arm radius and other design factors on the maximum output speed and velocity constant of a single-stage hydraulic servo for guided missile use. Methods are given for determining the best relay torque-arm radius, which may be applied to valves having non-linear flow and reaction characteristics.—(18.2).

MATHEMATICS

Elliptic functions and integrals with real modulus in fluid mechanics. R. Legendre. N.A.C.A. T.M. 1435. *June* 1958.

The usefulness of the Schwarz transformation and elliptic functions in defining flows outside and inside polygons is discussed and illustrated. Applications of the various functions are made to the calculation of wing profiles, of compressor profiles, and to the study of the vibration of aeroplane wings and of compressor vanes.—(22.1 × 1.10.1.1 × 1.5.2 × 2).

MECHANICAL ENGINEERING

The influence of plain bearing on shaft whirling. E. Downham. R. & M. No. 3046. 1958.

Experiments are described which show the effects of bearing length, bearing clearance, and lubricant viscosity on the critical whirling speeds of a single-shaft rotor system supported in plain bearings.—(23 × 27).

Results of an experimental investigation of small viscous dampers. M. A. Silveira et al. N.A.C.A. T.N. 4257. *June* 1958.

Experimental results on the damping characteristics of small viscous dampers are presented. Several types of fluid-displacement and viscous-shear dampers for varying physical and fluid properties were tested. Descriptions of test devices are given.—(23 × 1.12).

METEOROLOGY

The effect of icing on aircraft equipped with vortex generators. A. D. Wood and D. G. Gould. N.A.E. Report LR-220. April 1958.

By reproducing in plastic the ice formation which has been found to build up on vortex generators tested under heavy icing conditions in a low temperature wind tunnel, it has been possible to carry out flight tests to determine the effects of such icing on vortex generator installations on a Harvard and on a CF-100 aircraft.—(24).

MISSILES

Application of statistical theory to beam-rider guidance in the presence of noise. II—Modified Wiener filter theory. E. C. Stewart. N.A.C.A. T.N. 4278. *June* 1958.
An application has been made of Newton's modification of Wiener filter theory to optimise a beam-rider guidance system operating in the presence of glint noise. The system transfer functions and error performance against noise are given. Requirements on control deflection rates, circuit voltages and servo energy are discussed.—(25.1).

POWER PLANTS

See also MECHANICAL ENGINEERING

The icing of compressor blades and their protection by surface heating. G. L. Shires and G. E. Munns. R. & M. 3041. 1958.
An investigation into the feasibility of protecting compressor blades against ice accretion by surface heating is described. An electrically heated blade, having almost uniform surface temperature was used to determine the effects of air velocity and icing conditions on the minimum heat flow required to prevent the formation of ice. Gas-heated blades with various internal passage shapes were also tested. A method of estimating the pressure drop through the gas-heated blades is suggested and a worked example is included.—(27.1×18.4).

Experimental investigation of an impulse-type supersonic compressor rotor having a turning of 73° *at the mean radius. J. R. Sterrett. N.A.C.A. T.N.* 4252. *June* 1958.
An impulse rotor with guide vanes was designed and tested in Freon-12. The rotor had an inlet and exit hub-tip ratio of 0·575 and 0·75, respectively. The blade-passage design was based on a two-dimensional vortex-flow theory modified to account partially for three-dimensional compressions.—(27.1× 1.5.2).

PROPELLERS

See AERODYNAMICS—INTERNAL FLOW

REFERENCE LITERATURE

See AERODYNAMICS—THERMO-AERODYNAMICS

FATIGUE

Further investigation of fatigue-crack propagation in aluminum-alloy box beams. H. F. Hardrath and H. A. Leybold. N.A.C.A. T.N. 4246. *June* 1958.
Results are presented for several designs of built-up box beams tested at one load level. Stress redistribution during the tests was studied to aid in interpreting the results. Parameters which varied were the material of construction, mode of connecting stringers to cover skins, ratio of stringer area to skin area, and location of rivets with respect to line of crack propagation.—(31.2.3.3.1).

SCIENCE GENERAL

See also AERODYNAMICS—TESTING AND INSTRUMENTS

Reflection and transmission of sound by a slotted wall separating two moving fluid streams. R. L. Barger. N.A.C.A. T.N. 4295. *June* 1958.
The acoustics problem discussed is related to the aerodynamic problem of determining the tunnel-wall interference on an oscillating aerofoil in a slotted-throat wind tunnel in that the same boundary condition is involved with one of the two streams at the boundary having zero velocity. In the analysis the wall with discrete slots is replaced by an equivalent homogeneous boundary.—(32.2.3×1.12.1.2).

Far noise field of air jets and jet engines. E. E. Callaghan and W. D. Coles. N.A.C.A. Report 1329. 1957. *Supersedes N.A.C.A. T.N.* 3590 *and* 3591.
The effect of nozzle-exit shape was studied using a small air jet. Circular, square, rectangular and elliptic convergent nozzles and convergent-divergent and plug nozzles were investigated. The acoustic power radiated by the engines and air jet was correlated by the Lighthill parameter. The free-field spectral distribution of the sound power was in good agreement for both engine and air jet.—(32.2.3).

Note on acoustic energy flow in a moving medium. H. S. Ribner. U.T.I.A. T.N. 21. *April* 1958.
Both acoustic energy density and energy flow are known to be modified by motion of the medium. Comparison is made of the formulae of three investigators. Examples show how variations in the velocity of a stream carrying plane sound waves can change the "linear theory" acoustic energy density from positive through zero to negative, with corresponding changes in the energy flow.—(32.2.3).

On the strength distribution of noise sources along a jet. H. S. Ribner. U.T.I.A. Report 51. *April* 1958.
The spatial distribution of noise sources along a jet is investigated theoretically. The analysis refers to the noise power emitted by a "slice" of jet as a function of distance of the slice from the nozzle. Because of the striking attenuation of strength with distance, the "fat" part of the jet must contribute much less to the total noise power than is commonly supposed. Further implications, especially for multiple-nozzle and corrugated mufflers, are discussed.—(32.2.3).

Stabilisation des transports d'énergie a longue distance. R. Perret. Pubs. Sc. et Techs. NT. 75. 1958. (*In French*).
A theoretical investigation of small oscillations in electrical generators connected by multiphase lines is given. Application to the analagous study of the artificial synchronisation of an alternator connected to a network by a very long line is also given.—(32.3.1).

Optique de la photographie sous-marine. C. Maureau. Pubs. Sc. et Techs. NT. 74. 1958. (*In French*).—(32.2.4).

STRUCTURES

LOADS

Flight and analytical methods for determining the coupled vibration response of tandem helicopters. J. E. Yeates et al. N.A.C.A. Report 1326. 1957. *Supersedes N.A.C.A. T.N.* 3849 *and* 3852.
A discussion of flight-test and analysis methods for some selected helicopter vibration studies is presented. The use of a mechanical shaker in flight to determine the structural response is reported. Some results are presented to show the coupled response of a helicopter in flight for two different blade configurations. A method is also presented for calculating the natural coupled frequencies and mode shapes of vertical vibrations of tandem helicopters; certain forced excitations are also considered. Frequency response functions of cockpit deflection for various applied oscillating forces are given.—(33.1.2×2×1.11).

THEORY AND ANALYSIS

Effect of a stringer on the stress concentration due to a crack in a thin sheet. J. L. Sanders. N.A.C.A. T.N. 4207. *March* 1958.
A coefficient is obtained for determining the effect of a reinforcing stringer on the stress concentration factor at the tip of a crack in a thin sheet. The results are given for the case in which the stringer is intact and for the case in which the stringer is broken. In the first case the stress concentration factor for the stringer is also given.—(33.2.4.13).

On solutions for the transient response of beams. R. W. Leonard. N.A.C.A. T.N. 4244. *June* 1958.
Williams' type modal solutions of the elementary and Timoshenko beam equations are presented for the response of several uniform beams to a general applied load. Example computations are shown for a free-free beam subject to various concentrated loads at its centre. Discussion includes factors influencing the convergence of modal solutions and factors to be considered in a choice of beam theory. Results obtained by the travelling-wave method and Houbolt's method are also presented and discussed.—(33.2.4.1.10).

An empirical method of determining the crippling stress for flanged frames of Alclad 24*S-T at room temperature and elevated temperature. G. Anevi. S.A.A.B. T.N.* 40. *December* 1957.
A series of tests to determine experimentally the crippling stress for a particular range of straight and curved flanged frames of Alclad 24S-T at room and elevated temperatures is described. Based on the results of these tests, an empirical method is suggested for the determination of crippling stress.—(33.2.4.4.4).

The finite twisting and bending of heated elastic lifting surfaces. R. L. Bisplinghoff. Inst. Flug. und Leich. Zurich. Mitt. Nr. 4. *(In English.)*

An investigation is described, and results are presented, on the topic of the finite twisting and bending of heated elastic lifting surfaces. The investigation encompasses both theory and experiment and its scope is confined to very long elastic surfaces of rectangular plan form. The problems investigated are therefore essentially of a two-dimensional nature. Special attention is paid to the character of the chordwise deformations and the role played by these deformations in coupling the bending and twisting actions of the surface. The theoretical approach is based upon the shallow shell theory of Marguerre, modified to include stiffened shells with temperature gradients over the surface and throughout the thickness.—(33.2.3.1 × 2).

On the post-buckling behaviour of stiffened plane sheet under shear. E. H. Mansfield. R. & M. 3073. 1958.

The post-buckling behaviour of a flat plate reinforced by stringers and frames is considered theoretically, attention being concentrated in the case of pure shear loading. Formulae and graphs are presented for the rapid determination of the shear stiffness, shear strain and induced compressive stresses in stringers and frames. The analysis is an extension of work by Kromm and Marguerre.—(33.2.4.6.4).

Il metodo delle forze bilanciate nella scienza e nella tecnica. L. Broglio. Mono. Sc. Aero. Nr. 3. *December* 1957. *(In Italian.)*

The basic concepts of the "Balance Method" when applied to problems of Applied Mechanics is illustrated, with special emphasis for structural problems. In Part I the aim and the technique of the method are described in a general form. Part II shows a detailed application of the method to typical structures, to show how the technique is to be applied for practical purposes. In Part III some numerical examples are developed, which concern the structures treated in Part II.—(33.2).

TESTING

Strain gage calibrations and flight loads testing techniques. N. A. Krause. AGARD Report 113. *April-May* 1957.

Methods of installation and calibration of strain gauge systems are discussed. Flight manoeuvres and methods of execution to obtain the desired data are outlined and typical test data shown. Techniques used in the handling and analysis of the data are discussed. The use of automatic electronic equipment and other development programmes which are improving the efficiency of conducting loads programmes are discussed.—(33.3).

THERMODYNAMICS

Étude des flammes stabilisées par des obstacles de révolution dans des écoulements subsoniques a grandes vitesses. Structure thermo-chimique du noyau de stabilisation. J. Rappeneau et H. Grassin. O.N.E.R.A. N.T. 41. 1957. *(In French.)*

Flames stabilised by bodies of revolution were studied qualitatively by visualisation techniques and quantitatively by analysing the gas (by determining the quantity of CO and CO_2).—(34.1).

Applications des constantes et données thermodynamiques des mélanges gazeux températures élevées. (Flammes, Moteurs et Propulseurs.) G. Ribaud et N. Manson. Pubs. Sc. et Tech. No. 341. 1958. *(In French.)*

By means of three numerical examples the application of the thermodynamic constants and data of gases and their mixtures at high temperatures are shown. The examples give details of the methods of calculations, using thermodynamic diagrams, of the characteristic states of the evolution of the working fluid in combustion chambers, in reaction engines (turbo-jets, ram-jets, rockets), internal combustion engines, and behind shock waves and explosive waves.—(34.4).

Approximations for the thermodynamic and transport properties of high-temperature air. C. F Hansen. N.A.C.A. T.N. 4150. *March* 1958.

Approximations are presented for the thermodynamic and transport properties of high-temperature air. Compressibility, energy, entropy, the specific heats, the speed of sound, the coefficients of viscosity and thermal conductivity, and the Prandtl number for air are tabulated from 500° to 15,000°K over a range of pressures from 0·0001 to 100 atmospheres.—(34.1 × 1.9.1).

Composition and thermodynamic properties of air in chemical equilibrium. W. E. Moeckel and K. C. Weston. N.A.C.A. T.N. 4265. *April* 1958.

Charts have been prepared relating the thermodynamic properties of air in chemical equilibrium for temperatures to 15,000°K and for pressures from 10^{-5} to 10^{+4} atmospheres. Also included are charts showing the composition of air, the isentropic exponent, and the speed of sound.—(34.1).

Studies of OH, CO, CH and C_2 radiation from laminar and turbulent propane-air and ethylene-air flames. T. P. Clark. N.A.C.A. T.N. 4266. *June* 1958.

OH, CO, CH and C_2 radiations from propane-air and ethylene-air flames were isolated with a monochromator, and the variation of emitter intensity with equivalence ratio was determined.—(34.1.1).

On pairs of solutions of a class of internal viscous flow problems with body forces. S. Ostrach and L. U. Albers. N.A.C.A. T.N. 4273. *June* 1958.

In previous analyses of fully developed combined forced- and natural-convection flows, examples were presented of two distinct states of flow and heat transfer which were obtained for a given set of conditions if the frictional heating was taken into account. These pairs of solutions are discussed in greater detail and show how the solutions are affected by systematic variations of the basic physical parameters.—(34.3.2 × 1.9.1).

Free convection under the conditions of the internal problem. G. A. Ostroumov. N.A.C.A. T.M. 1407. *April* 1958.

Mathematical theory of steady and unsteady gravitational convection is presented. General non-linear equations that can be linearised are emphasised. A discussion of the nonlinear theory is presented. Experimental investigations of phenomena described by both the linear and non-linear equations are considered, and experimental techniques are discussed. One chapter is concerned with the characteristics of water as the working fluid. Tabulated data are given on the thermal and convective parameters of water and are compared with those for other fluids.—(34.3.2 × 1.9.1).

Basic considerations in the combustion of hydrocarbon fuels with air. Lewis Flight Propulsion Laboratory. N.A.C.A. Report 1300. 1957.

Basic combustion research is collected, collated, and interpreted as it applies to flight propulsion. The following fundamental processes are treated in separate chapters: atomisation and evaporation of liquid fuels, flow and mixing processes in combustion chambers, ignition and flammability of hydrocarbon fuels, laminar flame propagation, turbulent flames, flame stabilisation, diffusion flames, oscillations in combustors, and smoke and coke formation in the combustion of hydrocarbon-air mixtures. Theoretical background, basic experimental data, and practical significance to flight propulsion are presented.—(34.1).

Relation of turbojet and ramjet combustion efficiency to second-order reaction kinetics and fundamental flame speed. J. H. Childs et al. N.A.C.A. Report 1334. 1957.

Theoretical studies of jet engine combustion processes are summarised and the equations are applied to experimental data.—(34.1.1).

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